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Integrated Digital/Electric Aircraft Concepts Study

M. J. Cronin, A. P. Hays,
F. B. Green, N. A. Radovcich,
C. W. Helsley, and W. L. Rutchik

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Burbank, California

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FOREWORD

This study was conducted under NASA contract NAS1-17529 by the staff of the Lockheed-California Company and Lockheed-Georgia Company, divisions of the Aeronautical Systems Group of the Lockheed Corporation.

NASA's Technical Representative was C. R. Spitzer of the Flight Control Systems Division of Langley Research Center. His guidance and advice is greatly appreciated.

At Lockheed, the study was under the supervision of J. Lynn, Department Manager for Advanced Design of Land-Based Aircraft (Lockheed-California Company) and D. B. Mulcare (Lockheed-Georgia Company). The program manager was A. P. Hays and engineering manager was M. J. Cronin.

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Information on subsystems definition, performance, costs, and weights were also received from

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NOTE: The use of trade names and descriptions of these products are for the purposes of definition only. They do not indicate approval or endorsement by NASA.

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INTEGRATED DIGITAL/ELECTRIC AIRCRAFT CONCEPTS STUDY

M. J. Cronin, A. P. Hays, F. B. Green,
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SUMMARY

The Integrated Digital/Electric Aircraft (IDEA) is an aircraft concept which employs the following systems: Advanced Electrical Power Generation and Engine Starting, Power Distribution and Management, Electromechanical Actuation Systems, Electrical Deicing, Electrically-Driven Environmental Control System, Advanced Data Distribution, and Digital Flight Control System. After detailed trade analysis of candidate systems, preferred systems were applied to the baseline configuration, which contains advanced technology in all areas except for secondary power and flight control systems. The Baseline is a stretched-fuselage Lockheed L-1011 derivative carrying 350 passengers over a design range of 4600 n.mi. with performance benefits evaluated at the average stage length of 2500 n.mi. It was assumed that the advanced technologies would reach "technology readiness" (be at a sufficiently advanced stage of development for commitment to an aircraft design) in 1990 for an aircraft entering service in 1994. An additional configuration, the Alternate IDEA, was also considered. For this concept the design ground rules were relaxed in order to quantify additional synergistic benefits.

The major objectives of the study were to define an IDEA configuration, define technical risks associated with the IDEA systems concepts, and to identify the research and development required activities to reduce these risks.

Following a trade-off of candidate subsystems, the selected subsystems include: power generation - two samarium cobalt starter/generators per engine, producing variable voltage/variable frequency with selective conversion (to 270 Vdc or to constant voltage constant frequency); power distribution - hybrid ring/distributed bus; actuators - rotary power hinge operating through an antijam mechanism; environmental control system - electrically driven vapor cycle system and air compressor; flight controls systems - centralized/mesh architecture, (system reliability permits -10 percent static stability margin).

Benefits were determined using an aircraft performance and sizing computer program. When the IDEA concepts were incorporated into the design and the aircraft resized for the design mission, block fuel was predicted to decrease by 11.3 percent, with 7.9 percent decrease in direct operating cost. The Alternate IDEA shown a further 3.4 percent reduction in block fuel and 3.1 percent reduction in direct operating cost, primarily as a result of reduction in maximum limit maneuver load factor from 2.5 to 2.0.

Plans show that technical risks exist in power generation and motor technology (primarily associated with higher operating temperatures); power distribution (fault sensing, clearing and survivability); electromechanical

actuators (cooling and antijam design); power controllers (adequate capacity and cooling) flight control system (demonstration of reliability of flight-critical system). In these technical areas government research would be beneficial in reducing these risks.

INTRODUCTION

Designers of commercial aircraft strive to provide transportation at the lowest possible cost consistent with such constraints as safety, reliability, and comfort. Aircraft operating cost is usually measured as the sum of direct operating cost (DOC), usually defined by the Air Transportation Association of America (ATA) method, and indirect operating cost, which involves such factors as passenger service and ground handling costs. DOC is therefore of more interest to the designer, and it consists of the sum of four major terms with approximate contribution as follows:

1.	Flight Crew	15%
2.	Fuel and Oil	45%
3.	Depreciation	20%
4.	Maintenance	20%

In addition, insurance is about 1 percent of DOC. The cost of borrowing money is not included.

DOC can therefore be reduced by reducing aircraft fuel burned, purchase price, and maintenance costs. Both fuel burned and aircraft empty weight (which is closely related to purchase price) can be reduced through reduction of three aircraft and engine parameters - 1) aircraft drag, 2) aircraft weight, and 3) engine specific fuel consumption (SFC).

NASA has had a leading role in the sponsorship of research and technology development to reduce operating costs. One avenue of sponsorship has concerned the application of advanced electric and electronic technologies to commercial aircraft. The Lockheed-California Company has participated in two previous studies; the first was a study funded by the Johnson Space Center entitled "Application of Advanced Electric/Electronic Technologies to Conventional Aircraft" and completed in 1981 (ref. 1). The objective of the study was to evaluate the benefits of advanced electric/electronic systems, as used in the Space Shuttle, to commercial aircraft. Building on this study, a second study was performed under funding from Langley Research Center. The title of this study is "Electronic/Electric Technology Benefits Study" (ref. 2). This study qualitatively evaluated 135 separate subsystems concepts, from concepts such as the application of adaptive controls, to hardware such as powered wheels on landing gear. Of these, eight systems-oriented concepts were quantitatively evaluated:

- Near-term flight controls (RSS, not flight critical)
- Far-term flight controls (RSS, flight critical)

- Near-term secondary power (No engine bleed)
- Far-term secondary power (No bleed, no hydraulics)
- Advanced avionic components (Advanced electronics and packaging)
- Advanced cockpit (Controls and multipurpose displays)
- Air traffic control (Onboard and ATC system techniques)
- Advanced systems aircraft (All of the above technologies)

The Integrated Digital/Electric Aircraft (IDEA) concept follows these prior studies to incorporate two separate but related areas of technology. The first area of technology is the "all-electric aircraft" concept, in which aircraft hydraulic and pneumatic subsystems are replaced with electric actuators, electrically driven engine starters and environmental control system (ECS). There is a significant net weight reduction from the elimination of hydraulic and pneumatic hardware and ducts, and a reduction in engine SFC resulting from the substitution of compressor bleed with electrically driven compressors. The second is the application of digital control to aircraft flight control, and other systems. This results in more sophisticated control systems, which at the same time have levels of reliability comparable to that of the aircraft structure itself. These control systems permit active controls, such as relaxed longitudinal static stability and flutter mode suppression, which result in both weight savings and drag reduction.

The IDEA concept promises reductions in aircraft drag, empty weight, and engine SFC, resulting in significant reductions in DOC.

Objectives

There are three major objectives. The first is the quantification of benefits of the IDEA concepts to a transport aircraft entering service in the early 1990's time period, (the representative aircraft selected by Lockheed is a 350-passenger derivative of the L-1011 with a range of 4600 n.mi.) and to determine whether individual or collective portions of the IDEA concepts show promise for new or unique transport configuration designs not considered feasible until now. The second objective is to define associated risks associated with the application of this technology. The third is to identify the research and development to reduce these risks for potential application to transport aircraft.

Quantification of benefits. - In order to quantify the benefits of IDEA concepts the following subsystems are applied to an advanced technology commercial aircraft:

digital flight controls (primary and secondary)

digital data distribution

electromechanical actuation systems

advanced power generation systems

electric power distribution

electric power management

electric engine starting

electric wing and engine ice protection

electric environmental control system

New transport configuration design. - In quantifying the benefits of the IDEA concepts, the aircraft design was held constant, with the exception of resizing the wing and engine for wing loading and thrust/weight ratio to meet performance constraints, and resizing the horizontal tail for relaxed longitudinal stability. An additional objective therefore is to determine whether further benefits accrue if the IDEA concepts permit new or unique transport configuration designs. The objective is embodied in the "Alternate IDEA Based Configuration," which has an undefined service entry date, and for which fundamental changes in aircraft design are permissible.

Technology development plans. - Technologies and methodologies are identified which must be developed and validated by 1990 to achieve the IDEA based configuration design.

In addition, a subset of these technologies and methodologies are identified which are consistent for government support consistent with the guidelines of longterm, high risk, and high payoff. Resources and schedule requirements for this subset are identified.

Approach

Method of evaluating benefits. - The benefits are evaluated by first defining analytically a baseline aircraft which incorporates advanced technologies (composites, energy efficient engine, supercritical wing, and active ailerons) in all areas except for the subsystems to be evaluated. These subsystems are removed from the aircraft and are replaced with those incorporating the IDEA concepts. The subsystems to be incorporated result from trade-offs of candidate subsystems based on weight, cost, reliability, maintainability, compatibility with other subsystems, and other factors that may affect selection of candidates.

A detailed evaluation is made of the preferred subsystems in order to make a quantitative evaluation of the benefits of the IDEA concepts in terms of aircraft block fuel and DOC. This evaluation is carried out using the Lockheed Advanced Systems Synthesis and Evaluation Technique (ASSET) computer program which permits changes in aircraft component weights or performance to

be incorporated; the aircraft is analytically flown through its required mission and changes in aircraft weight, fuel, and cost readily determined.

The Alternate IDEA Based Configuration is evaluated in a similar manner, but with the configuration design constraints removed. Because the service entry date is undefined, other IDEA concepts may be added that do not meet the requirements for an entry into service in the early 1990s.

Technology development plans and risks. - Technology development plans are established which will reduce technology risks to an acceptable level. These plans were developed in cooperation with manufacturers of subsystem equipment to reflect the best estimate of resources required to bring the technologies to a state of technology readiness.

For technologies to be incorporated into the IDEA based configuration, the technology development schedule is assumed which is illustrated in figure 1. Technology validation will be achieved by the end of 1990, leading to FAA Certification and entry into service in 1994.

For the Alternate IDEA Configuration it is assumed that the certification date is undefined.

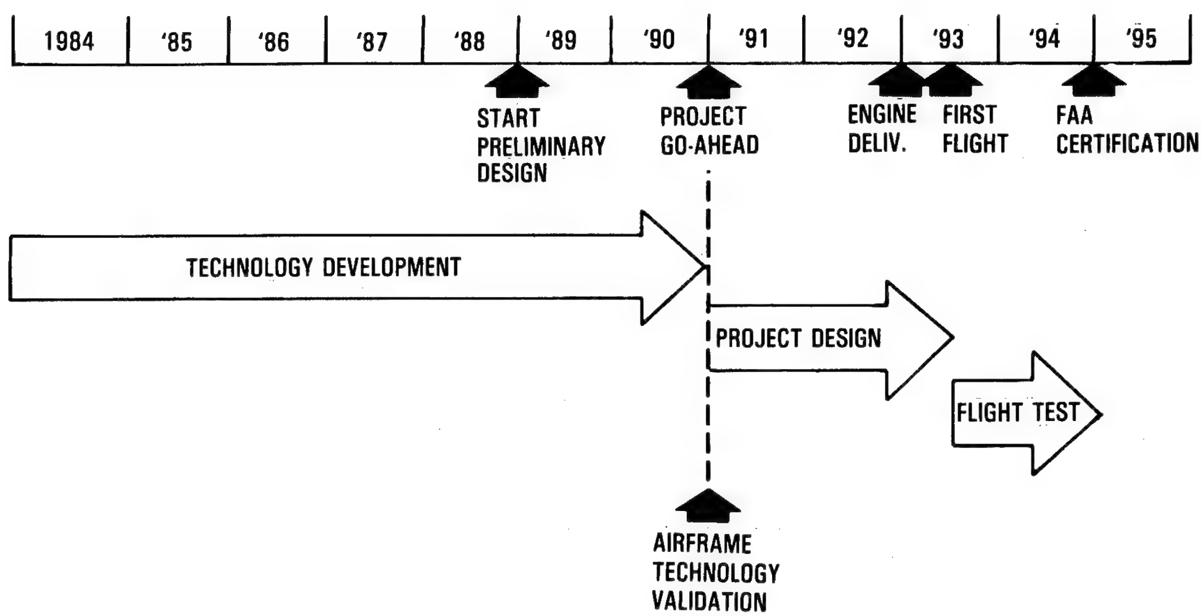


Figure 1. - Development schedule.

ABBREVIATIONS AND ACRONYMS

AAES	advanced aircraft electric system
ac	alternating current
ACARS	ARINC communications, addressing, and reporting system
ACB	alternating current bus
ACES	automatic controlled electric system
ACM	air cycle machinery
ACS	active control system
Act	actuator
A/D	analogue to digital
AEECS	all electric environmental control system
ADEMS	advanced display and electrical management system
Adv	advanced
AEDS	advanced electric distribution system
AFCM	advanced flight control manager
AFMS	automatic flight management system
AFWAL	Air Force Wright Aeronautical Laboratories
AGB	accessory gear box
AGD	axial gear differential
AH	ampere-hour
AIDS	aircraft integrated data system
AIPS	advanced information processing system
AIRLAB	Avionics Integration Research Laboratory
ALMS	automatic load management system, electrical
APGS	advanced power generation system
A/P	autopilot

APU	auxiliary power unit
APUC	APU contactor
ARINC	Aeronautical Radio Incorporated
ASSET	advanced systems synthesis and evaluation technique
A/T	autothrottle
ATA	Air Transport Association
ATM	air turbine motor
BC	bus controller
BCAS	beacon collision avoidance system
BIT	built-in test
BITE	built-in test equipment
BP	bus power
BPCU	bus control power unit
BPV	bypass valve
BT	bus tie
BTC	bus tie contactor
C	contactor
CARSRA	computer-aided redundant system reliability analysis
CB	circuit breaker
CC	cabin compressor
CDU	control and display unit
cfm	cubic feet per minute
c.g.	center of gravity
CITS	central integrated test systems
COMP	compressor
COND	condenser

CRAD	contractor research and development
CRT	cathode ray tube, central gun type
csd	constant speed drive
CSDL	Charles Stark Draper Laboratory
CT	current transformer
CTD	control technology demonstrator
CTFE	chlorotrifluouroethylene
CV	check valve
CVCF	constant voltage constant frequency
D/A	digital to analogue
DADS	digital air data systems
DAIS	digital avionics information system
DEMUX	demultiplexer
dc	direct current
DFCS	digital flight control system
DHS	data handling system
DITS	digital information transfer system
DLC	direct lift control
DME	distance measuring equipment
DoD	Department of Defense
DOC	Direct Operating Cost
DPCT	differential protection current transformer
DP/LM	digital power/load management
EAROM	electrically alterable read only memory
ECS	environmental control system
ED	engine driven

EH	electro-hydraulic
EIDS	electro-impulse deicing system
EM	electric motor or electromechanical
EMA	electromechanical actuator
EMAS	electromechanical actuator system
EMI	electromagnetic inteference
EMP	empennage
EMS	electric management system
EMUX	electrical multiplexing
EPC	external power contactor
E/PROM	erasable/programmable read only memory
EPS	emergency power supply
EPU	emergency power unit
ESS	essential
E^3	energy efficient engine
FAA	Federal Aviation Administration
FAA AC	Federal Aviation Administration Advisory Circular
FACS	flight augmentation computer system
FADEC	full authority digital engine control
FAR	Federal Air Regulation
FBW	fly by wire
FC	flight computer
FC	freon compressor
FCC	flat conductor cable
FCC	flight control computer
FCCS	flight control computer system

FCS	flight control system
FDIR	fault detection, isolation, and recovery
FDM	frequency division multiplexing
FET	field effect transistor
FIP	fault indicator panel
FMCS	flight management computer system
FMS	flight management system
fpm	feet per minute
FS	flight station
FS	flight station loads
FSSB	flight station section breaker
FTC	fault-tolerant computer
FTMP	fault tolerant multiprocessor
FTP	fault tolerant processor
FW	full wave
FWC	flight warning computer
GATT	gate assisted turn-off thyristor
G/A	go-around
GEN	generator
GC	generator contactor
GCU	generator control unit
GPFC	general purpose flight computer
GTO	gate turn off
HF	high frequency
HMA	hydromechanical actuator
HOL	higher order language

HP	high pressure
HP	horsepower
HPX	horsepower extraction
HTR	heater (electric)
HX	heat exchanger
IAP	integrated actuator package
IDEA	integrated digital/electric aircraft
IDG	integrated drive generator
IDGS	integrated drive generator system
IEG/S	integrated engine generator/starter
IGV	inlet guide vanes
ILRU	independent line replaceable unit
ILS	instrument landing system
INS	inertial navigation system
INV	inverter, electrical
IOC	indirect operating cost
I/O	input/output
IOC	initial operational capability or indirect operating cost
IRAD	independent research and development
IRS	inertial reference system
ISEM	improved standard electric module
ITS	information transfer system
kips	thousand instructions per second
kvar	kilovar
kW	kilowatt
L/D	lift to drag ratio, aerodynamic

lb	pound
LCC	life cycle costs
LE	leading edge, aerodynamic
LED	light emitting diode
LaRC	Langley Research Center (NASA)
LeRC	Lewis Research Center (NASA)
LFC	laminar flow control
LMC	local management centers
LMLA	load management logic assembly
LRU	line replaceable unit
LRU	line replaceable unit, avionics
LSI	large scale integration, electronics
LVDT	linear variable displacement transformer
LVT	linear voltage transducer
LW	left wing
LWEL	left wing essential loads
LWL	left wing loads
LWSB	left wing section breaker
MAB	mid avionics bay
MAC	mean aerodynamic chord
Mach	speed of sound
MALMS	manned/automatic load management system
MBPS	megabits per second
MCU	master control unit
MDM	multiplex demultiplex (avionics unit)
MDT	multiplexer data terminal

M^2 FCS	multi-microprocessor flight control system
MELC	main electric load center
MINCOMS	multiple interior communications system
MLG	main landing gear
MPa	mega pascals, hydraulic pressure
MTBF	mean time between failures
MTTR	mean time to repair
MUX	multiplex
NADC	Naval Air Development Command
NASA	National Aeronautics and Space Administration
NAVAIR	Naval Air Systems
NON ESS	non essential
n.mi.	nautical miles
NRC	non-recurring cost
OAT	outside air temperature
OEW	operating empty weight
OF/UF	overfrequency/underfrequency
OV/UV	overvoltage/undervoltage
Omega	a hyperbolic navigation system
PACS	pitch active control system
PBW	power by wire
PC	power contactor
PC	power controller
PCP	pilot control panel
PDR	phase delay rectifier
PDU	power drive unit

PM	permanent magnet
PMS	performance management system
POR	point of regulation
ppm	pounds per minute
PRI	primary
psf	pounds per square foot
PPS	pounds per second
PRV	pressure regulating valve
psi	pounds per square inch
PSP	power system processor
PTO	power take-off
PTU	power transfer unit
PU	per unit
PWM	pulse width modulation
PWAM	pulse width amplitude modulation
PWT	programmable wiring terminal
QAD	quick attach/detach (device)
RA	radio altimiter
RAM	random access memory
RAT	ram air turbine
RC	recurring cost
RCCB	remote controlled circuit breaker
RCF	recirculation fan
R&D	research and development
RIT	remote input terminal
RF	radio frequency

RFI	radio frequency interference
RLC	remote load center
RLG	ring laser gyro
ROI	return on investment
ROT	remote output terminal
RPC	remote power controller
RSS	relaxed static stability or root sum square
RT	remote terminal
R&T	research and technology
RTCA	radio technical commission for aeronautics
RTU	remote terminal unit
RVDT	rotary variable displacement transformer
RW	right wing
RWEL	right wing essential loads
RWL	right wing loads
SAE	Society of Automotive Engineers
SAS	stability augmentation system
SCMP	self-checking multiprocessor pairs
SCR	silicon controlled rectifier
SEC	secondary
SEM	standard electronic module
SFC	specific fuel consumption
SIFT	software implemented fault tolerance
SmCo	samarium cobalt
SOC	spray oil cooled
SOSTEL	solid state electric logic

SPS	secondary power system
SPST	single pole single throw
SS	signal source
SSPC	solid state power controllers
TBO	time before overhaul
TDM	time division multiplexing
t/c	thickness/chord ratio
TL	tail loads
T/R	transmitter and receiver
TRU	transformer rectifier unit
TSB	tail section breaker
TSM	two speed motor
UART	universal asynchronons receiver/transmitter
VHF	very high frequency (radio set)
VHSIC	very high speed integrated circuits
VLSI	very large scale integration
VMOS	vertical metallic oxide semiconductor
VOR	VHF omni range, navigation
VOR	visual omni range
VSCF	variable speed constant frequency
VSD	vertical situation display
VVVF	variable voltage variable frequency
WRP	wing reference plane
4-D Nav	four dimensional navigation (3-D plus time)
ZIF	zero insertion force

1. GROUND RULES

1.1 Aircraft Design

Each aircraft is designed so that it would be expected to meet the requirements of FAR Part 25 and Part 36.

1.2 Design Requirements

Each configuration is sized based on achieving the design range with 100 percent load factor. Minimum DOC configurations are derived at the average stage length with 65 percent load factor.

1.2.1 Performance. -

- Cruise Mach No. = 0.8
- Still air design range on standard day = 4600 n.mi.
- Average stage length = 2500 n.mi.

1.2.2 Payload. -

- 350 passengers @ 165 lb/passenger = 57,750 lb
- Baggage = 15,750 lb
- Total payload = 73,500 lb
- Space-limit cargo (not included in performance calculations) = 31,600 lb

1.3 Performance Constraints

The following performance constraints are used:

- Takeoff field length = 10,500 ft @ S.L., 84°F
- Engine out second segment gradient = 2.7%
- Initial cruise altitude = 31,000 ft with climb capability of 3 ft/sec and 1.3g buffet
- Landing field length = 7,000 ft @ S.L., 84°F (L.W. = 302,000 lb)
- Approach Speed = 145 kt

- Required rate of climb at start of cruise, max. climb power = 180 ft/min

- Ratio of available wing fuel volume to required fuel volume = 1.1

1.4 Cost Assumptions

Direct operating costs (DOC) are calculated using the Air Transportation Association "Standard Method of Estimating Comparative Direct Operating Costs of Turbine Powered Transport Airplanes" with cost coefficients updated to 1983. All costs are based on constant 1983 dollars and a depreciation period of 16 years. Other assumptions associated with the calculation of DOC and return on investment (ROI) are shown in table 1.

1.5 Study Limitation

Some of the most significant limitations of this study are common to any study which attempts to forecast trends. For the most part the levels of

TABLE 1. FINANCIAL ASSUMPTIONS

Parameter	Value
Fuel Cost	$1.50/Gal
Depreciation Period	16 years
Salvage Value	10%
Maintenance Burden	3.4
Maintenance Labor Rate	$16.50/hour
(Total)/(Revenue) Flying Hours	1.004
Percent First Class	10%
Load Factor	65%
Manufacturers Profit Level	15%
Number of Aircraft - Development	5
- Production	295
Airline Fleet Size	23

technology assumed for advanced configurations are based on extrapolation of trends of technology advancements up to the present time, modified where necessary by judgments of specialists in each technological area. These predictions cannot account for fundamental breakthroughs which may (remotely) occur between now and technology readiness in 1990; however, it is rather unlikely that any major breakthrough could reach technology readiness by 1990. On the pessimistic side, these predictions cannot completely account for roadblocks in technology development. It is the purpose of risk assessment to make judgments about the probability and severity of roadblocks which may occur.

In assessing the benefits of these advances in technology by applying them to the reference aircraft, there are several cases where compromises in aircraft design must be made to implement these advances, and these compromises may involve many fundamental aspects of the design. Time and cost limitations preclude detailed trade-offs to find the best compromise, so that the net result is a configuration which is not completely optimized. The level of optimization reached is probably within about one or two percent of a perfectly optimized design, but we are looking for differences from the reference aircraft of the order of 5-10 percent by applying advanced technology elements, so that the one percent lost through failing to optimize the design may be significant.

In addition, aircraft design is a convergent iterative process. Preliminary estimates of, for example, c.g. range as a percentage of MAC, have proven to be slightly in error when the aircraft is resized. These errors do not significantly affect the quantification of benefits.

2. BASELINE AIRCRAFT

2.1 Origin of Configuration Design

The baseline configuration is a three-engined, wide-body commercial aircraft using the same engine layout as the Lockheed L-1011 TriStar. It also uses the same cockpit layout and fuselage diameter. Fuselage length is increased by 280 in as compared with the L-1011. In addition, the baseline includes a high aspect ratio wing with supercritical airfoil section, E^3 propulsion system, and graphite/epoxy composite materials.

Aircraft flight control and secondary power systems represent current technology, i.e., the level of technology as on the Boeing 757 and 767.

2.2 Power Generation System

In line with the philosophy of utilizing conventional systems in the baseline airplane, electric power in this airplane is derived from a constant-voltage constant-frequency (CV/CF) system using advanced configuration hydromechanical constant speed drives. These drives known as the Integrated Drive Generator System (IDGS) provide constant speed operation for the integrated generators and they offer attractive improvements over the

early axial gear differential (AGD) and the more recent integrated drive generator (IDG) configurations. The IDGS is more modular in design concept, and utilizes a side-by-side configuration (in which the generator is on a parallel axis with the hydromechanical log elements), and it has an improved hydraulic system. Physically, unlike the in-line design of the IDG, the IDGS is shorter in length and is a more compact, wider configuration that has a lower overhung moment on the engine pad. The modularity of the design also allows for more ready access to the power elements and, as a consequence, a reduction in mean-time-to-repair (MTTR). These physical design attributes along with the utilization of an advanced state-of-the-art microprocesser power management system make the IDGS an ideal candidate for the baseline airplane.

The elements of one IDGS power channel are shown in figure 2 to consist of a generator control unit (GCU), bus power control unit (BPCU) differential current transformers (CTs) and a quick attach/detach (QAD) (IDGS mount) adapter. The GCU is a primary component in that it controls and monitors the operational status of the drive-generator combination, the quality of the electric power and any exceedances from normal system limits (power characteristics, oil temperatures, etc.). As important protection features, the GCU provides overvoltage/undervoltage, overfrequency/ underfrequency, differential fault, open phase, and overload protection and it also detects the presence of component faults, within the IDGS, such as shorted diodes/PMG failures, etc. These latter protective monitoring functions are provided by built-in test

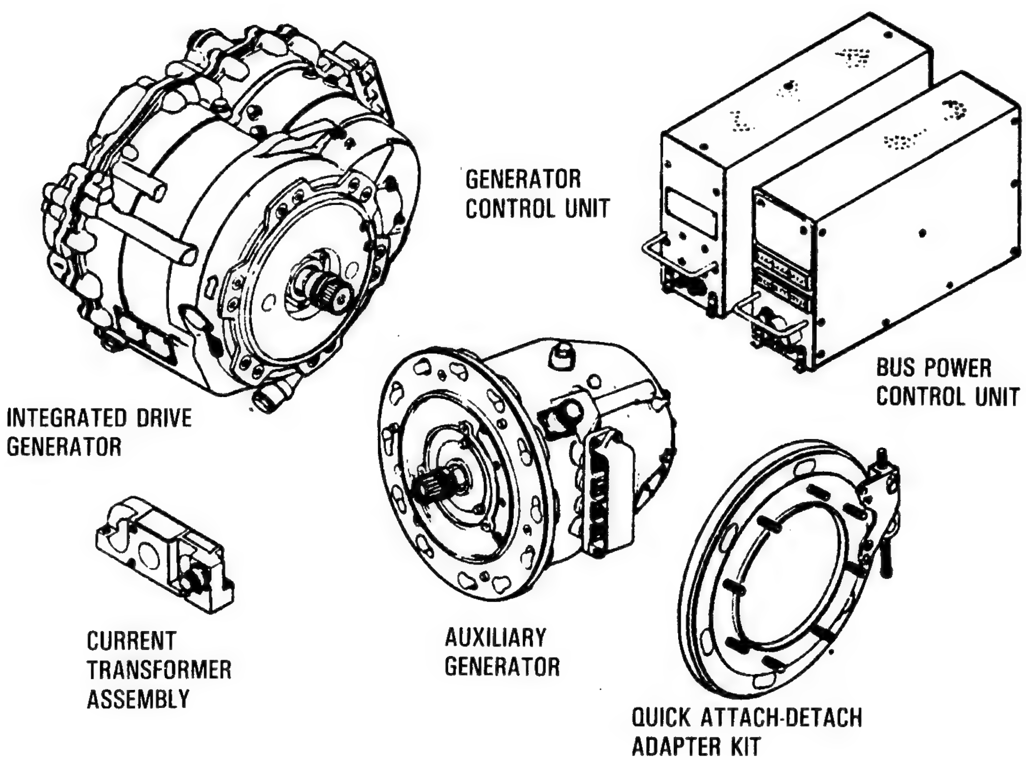


Figure 2. - IDGS power channel/APU generator (Courtesy Sundstrand Aviation).

equipment (BITE) which is integral with the GCU. Other refinements of the IDGS, as with the earlier IDG system, involve the use of line-current transformers (and other electric logic) to provide kilowatt (kW) and kilovar (kvar) load sharing, when the generator outputs are electrically paralleled. The GCU works closely with the BPCU and, through an intracommunication between these assemblies, the complete power system is controlled via the generator bus contactors (GBCs) and the bus tie contactors (BTCs).

Figure 3 is a schematic of the electric power system in the baseline airplane: it is designed as a three-channel paralleled system that can be operated as a paralleled or nonparalleled electric system. Each IGDS is rated as 90/120 kVA unit and the three generators are integrated with an onboard APU driven generator (of the same rating), and a ram air turbine (RAT) driven 5 kVA generator (the RAT also drives an emergency hydraulic pump unit). The design layout of the power system is different from the L-1011-500 configuration in that the use of the "go-around" emergency electric power system has been eliminated. In lieu of the latter, electric digital logic is used to drop nonessential electric loads off each of the three primary ac buses, leaving (in this emergency) only flight critical and other important loads (i.e., communications, etc.) connected to the buses. This emergency power configuration is considered an improvement over the "go-around" system in that it avoids the hazard of bringing power cables from each of the three generators into one area: it also acknowledges the remote possibility that bus faults can occur and it avoids the prospect of a sequential (cascade) failure of generators since essential/emergency loads are sourced by multiple buses rather than one.

As previously stated, the microprocessers in each power channel operate continuously to control and monitor the status of each power channel, but total power system management can be effected via the BPCUs. The BPCU is the master logic control (or co-ordination center) for the APGS (advanced power generation system) and as such it "manages" the power system, handles the communication with the GCUs, outputs BITE displays, and interrogates the non-volatile memories. Typically, the core of each GCU is a microprocessor, such as the Intel 8085, which operates with a 5 MHz clock: peripheral ports provide the interfaces between themicroprocessors and the logic circuit in the GCUs. The system also includes erasable/programmable read only memory (E/PROM), which provides the step-by-step instructions to the microprocessor; a nonvolatile memory (NVM), to store data in memory, without loss during power-OFF conditions and, an electrically-erasable read only memory (EAROM) to transmit data to the BPCU for display; also a random access memory (RAM), to handle port addresses, timer values, and sensor readouts. Other interfaces with the microprocesser control system includes A/D and D/A converters that communicates with the microprocessor via peripheral ports, as shown in figure 4. A universal asynchronous receiver/transmitter (UART) completes the system and this converts 8-bit serial data from the BPCU into parallel format for the microprocessor. Conversely, communication from the P to the BPCU involves parallel-to-serial conversion.

2.2.1 Emergency/APU ac power system. - Figure 5 is a simplified schematic of the low voltage 28 Vdc system which is powered from the primary ac power system. Three 28 V 200 A transformer rectifier units (TRUs) comprise the

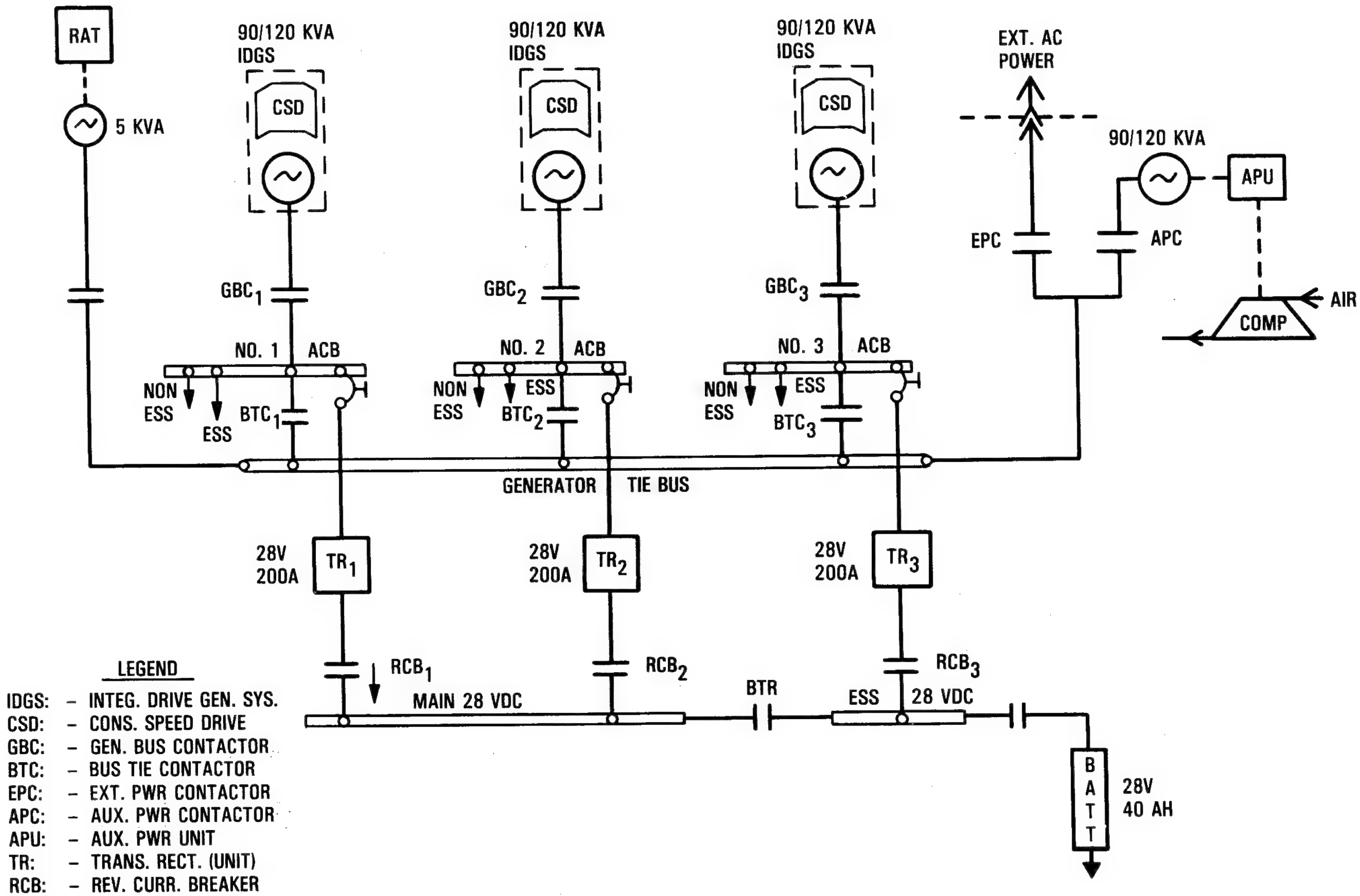


Figure 3. - Baseline electric power system.

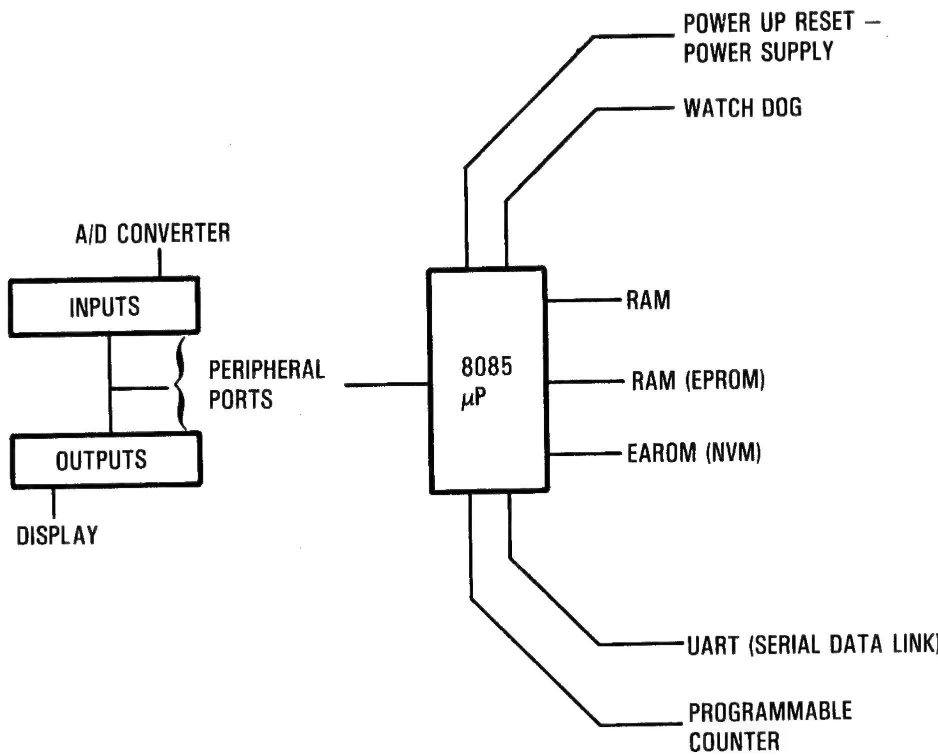


Figure 4. - Microprocessor Control System (Courtesy Sundstrand Aviation).

power supply for the aircraft's 28 Vdc loads: these are advanced state-of-the-art rectifier units which are self-cooled. A typical configuration of an emergency dc power system is as shown in figure 5A, in which the 28 Vdc emergency loads are shown connected to the emergency bus. The problem with this configuration is that a fault, on the emergency bus itself, could result in the loss of all essential and emergency dc loads. Figure 5B is therefore a preferred approach, since it allows the essential and emergency dc loads to be powered by multiple buses. As shown in the latter approach, the emergency battery is separately diode fed into the main and essential dc buses, with essential/emergency loads being connected to the two main dc buses. It would be possible also to use three dc buses (as with the ac system) and to use a tri-diode feed from the battery.

The baseline airplane does not impose the same critical concern on the ac power system as the IDEA since the flight surface controls are actuated hydraulically. As a consequence, the RAT drives an emergency pump unit which, in the event of an all engine-out condition, powers selected hydraulic jacks in each of the aircraft three primary control axis. This modus operandi is possible since, as described in Section 2.5, the inboard/outboard ailerons, the rudder and horizontal stabilizer all have redundant actuators that are mechanically summed onto each control surface. Isolation of some of the actuators is necessary to reduce the quiescent flow demand and to avoid overloading of the pump unit.

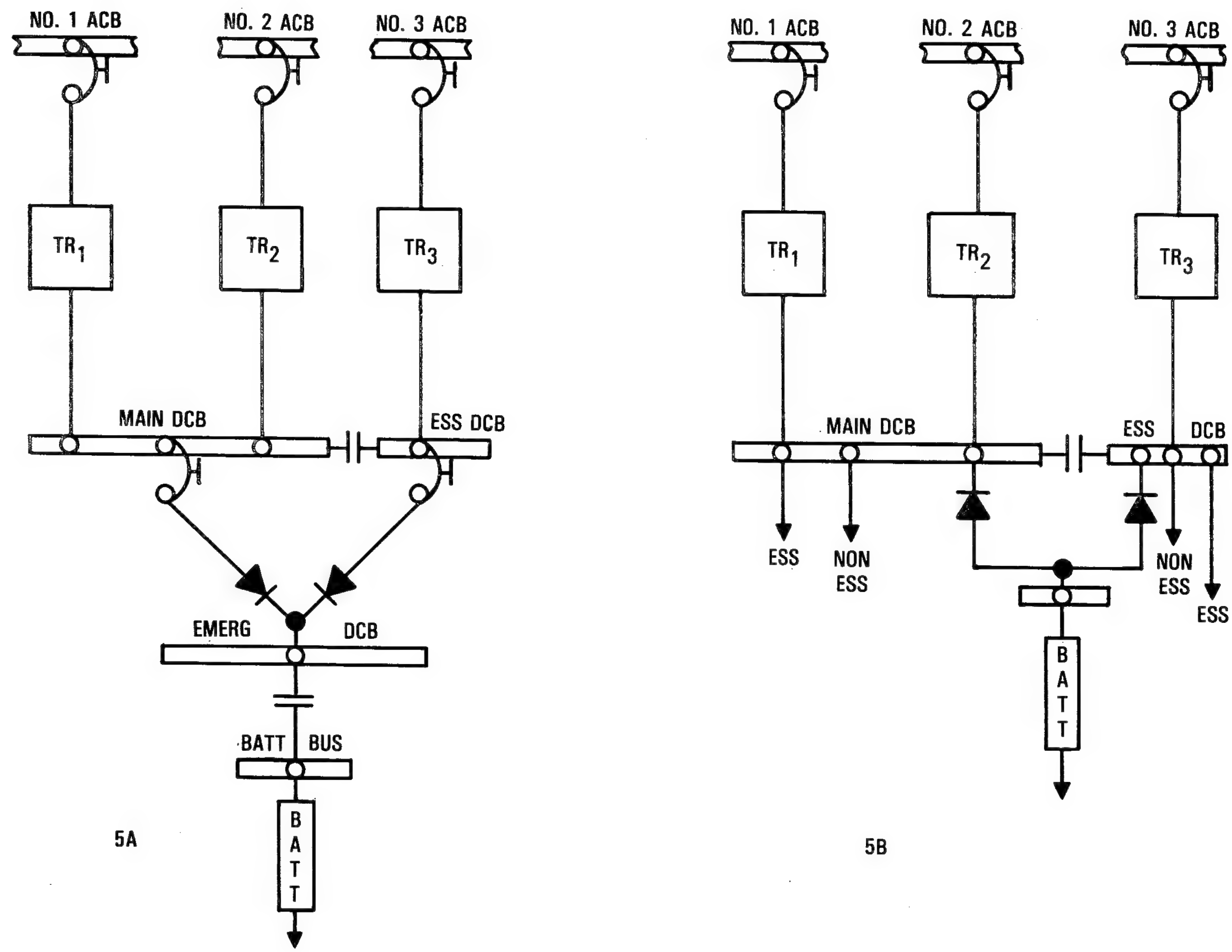


Figure 5. - 28 Vdc power system.

Because the baseline airplane uses a hydro-mechanical flight control system, the emergency electric power system is required only to support important instrumentation, engine/fuel controls and other miscellaneous electric and avionic functions. Fuel/oil shut-off valves, transfer valves, inlet doors, emergency lighting and certain displays/read-outs utilize 28 Vdc power, while synchros, motor driven fans, etc., utilize ac power from the RAT generator. Examination of figure 3 shows that the 28 V essential dc bus can be fed from the RAT generator via the No. 3 TRU. Under this mode of operation the bus tie contactor (BTC) will be open and nonessential loads on the main ac bus will be isolated.

The Baseline electrical power requirement is shown in Figure 6.

2.3 Hydraulic System

Supply pressures of 3000, 5000 and 8000 psi were considered for the baseline airplane. For high performance military aircraft, 5000 and 8000 psi systems have been espoused as offering prospectively lower weight, although this benefit is offset by higher acquisition costs and higher maintenance support costs, as reflected by a prospectively lower MTBF, lower reliability and increased leakage problems.

Rockwell-Columbus has been evaluating an 8000 psi lightweight hydraulic system for NADC and the SAE A-6 Subcommittee is constantly reviewing the progress on the 8000 psi system. Grumman Aircraft has also conducted a life cycle cost study on the F-14 showing a potential $297K/aircraft saving over the life of the aircraft. In addition ABEX and Vickers have been developing, testing and evaluating pumps for the 8000 psi system, using non-flammable fluids such as the A-02. Some early problems were identified with the pumps, but the hydraulic system suppliers are not identifying any unsurmountable problems.

As regards the installation aspects, components such as seals in actuators (and, particularly, high temperature/high pressure dynamic seals) are a subject for consideration and some concern: further, the concern for the reliability of production in-line joints and leakage leads to the adoption of the most scrupulous quality control methods for the welded and non-welded in-line joints. The higher specific weight of the fire-resistant fluids, such as the A-02 (chlorotrifluoroethylene (CTFE)), is an incentive to explore the higher pressure systems, but the efficacy of this high pressure system for the commercial airplane is debatable (particularly with the emphasis on DOCs). The normal hydraulic specifications have also relaxed (for the 8000 psi hydraulic system), the normal burst pressure limits of 3 to 5 times working pressures. Fluid velocities (depending on the line size) will also be higher; typically these will increase from the 15 to 30 fps to as high as 60 to 80 fps. In addition there are some practical performance considerations such as reduced bulk modulus and reduced stiffness of actuators in reacting high hinge moment loads. Finally there is the question as to the real weight savings in any aircraft where the ratio of small hydraulic actuators to large hydraulic actuators is large. It can be projected that the best weight benefits would spring (as with the high voltage electric system) from the transmission of

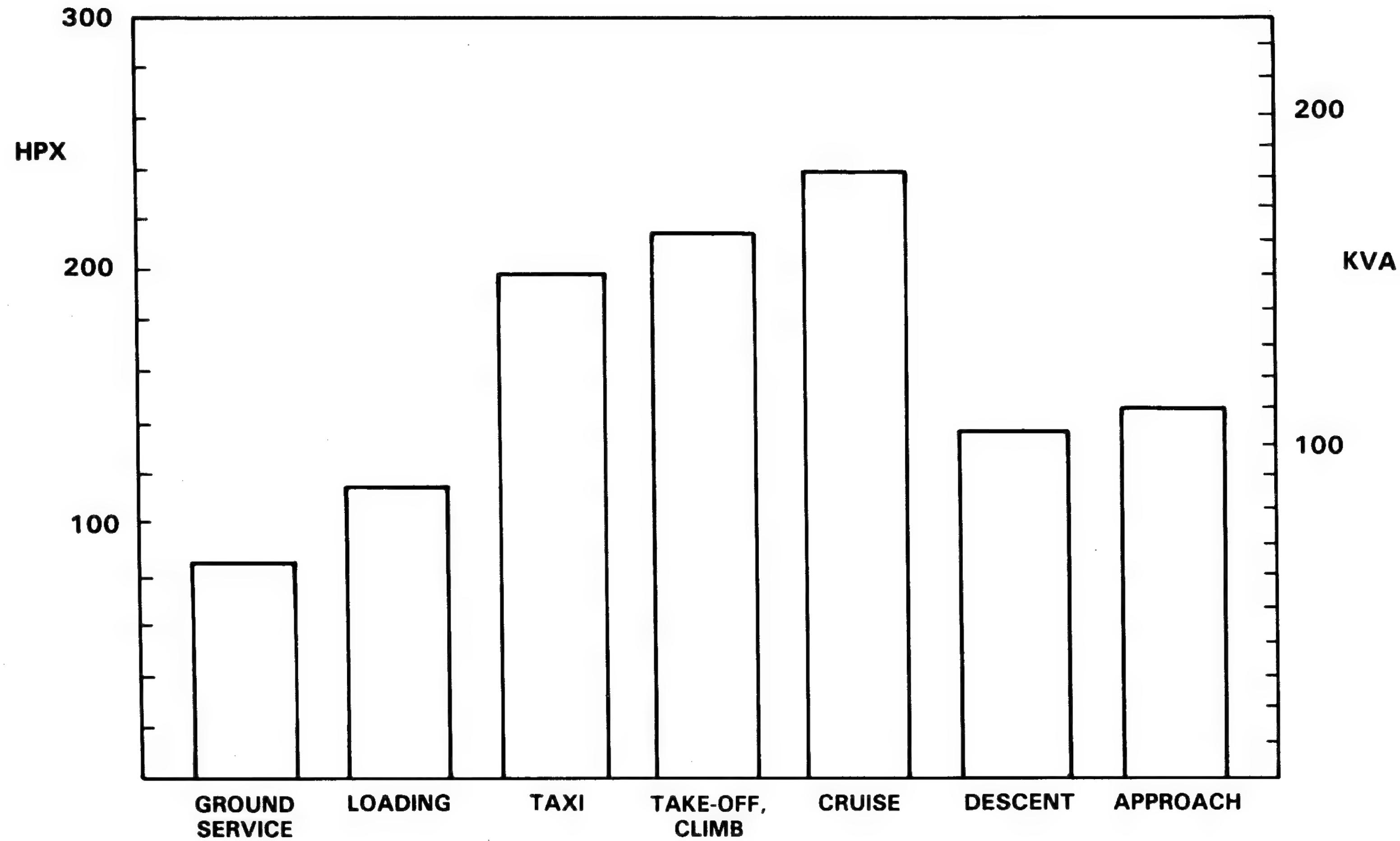


Figure 6. - Baseline electric power requirement.

high power over long distances. However, aside from landing gear and other large loads, etc., most actuators in the hydraulic system tend to be small (in their horsepower demand) and so line weight would not be reduced much with a higher pressure system. Another factor is that there is an unfavorable increase in the ratio of the tube wall thickness to the tube diameter in the low line sizes. A similar rationale applies to the actuator housings.

Based on the foregoing discussion, a conventional 3000 psi hydraulic system was selected for the Baseline airplane. This system comprises a total of four ED pumps: two on the No. 2 AGB and one on each of the Nos. 1 and 3 (engine) AGBs. Supporting the four ED pumps (to give additional flow capacity at takeoff and to furnish a high degree of redundancy/reliability are two ATM (air turbine motor) driven pumps and two electric (ac) motor driven pumps. To meet the all-engine-out condition and to provide emergency power for the FCS, a ram air turbine (RAT) driven pump is used to power selected critical FCS actuators.

Figure 7 is a simplified schematic of the hydraulic system: it is configured as a quad redundant system but this level of redundancy is not provided to all the aircraft loads. For example, the redundancy level of the hydraulic system behind the FCS actuators is as below.

Control Surface	Redundancy Level
Horizontal Stabilizer	4
Inboard Ailerons	3
Outboard Ailerons	2
Rudder	3
Spoilers	1

As shown in the hydraulic schematic, figure 6, the four systems A, B, C, D are individually supplied by the four ED pump units; however, two power transfer units (PTUs) interconnect systems A & B and C & D, without any interchange of fluid. The PTUs are back-to-back pumping elements that are reversible to the extent that one element can act as a motor, while the other acts as a pump. If, for example, there is a loss of pressure in system A (due to a No. 1 engine failure), pump B on No. 2 engine will operate the left side element of the PTU as a motor, and the RH side will operate as a pump to re-establish pressure in the A system. In addition to the PTUs, an ATM pump unit and an ac motor pump unit act as back-ups to each of systems B & C. Finally, the RAT pump unit ties into system D to meet the extreme emergency of the all-engine-out condition.

From the foregoing description, it can seem that the hydraulic system in the baseline airplane is configured as a highly-reliable/highly-redundant

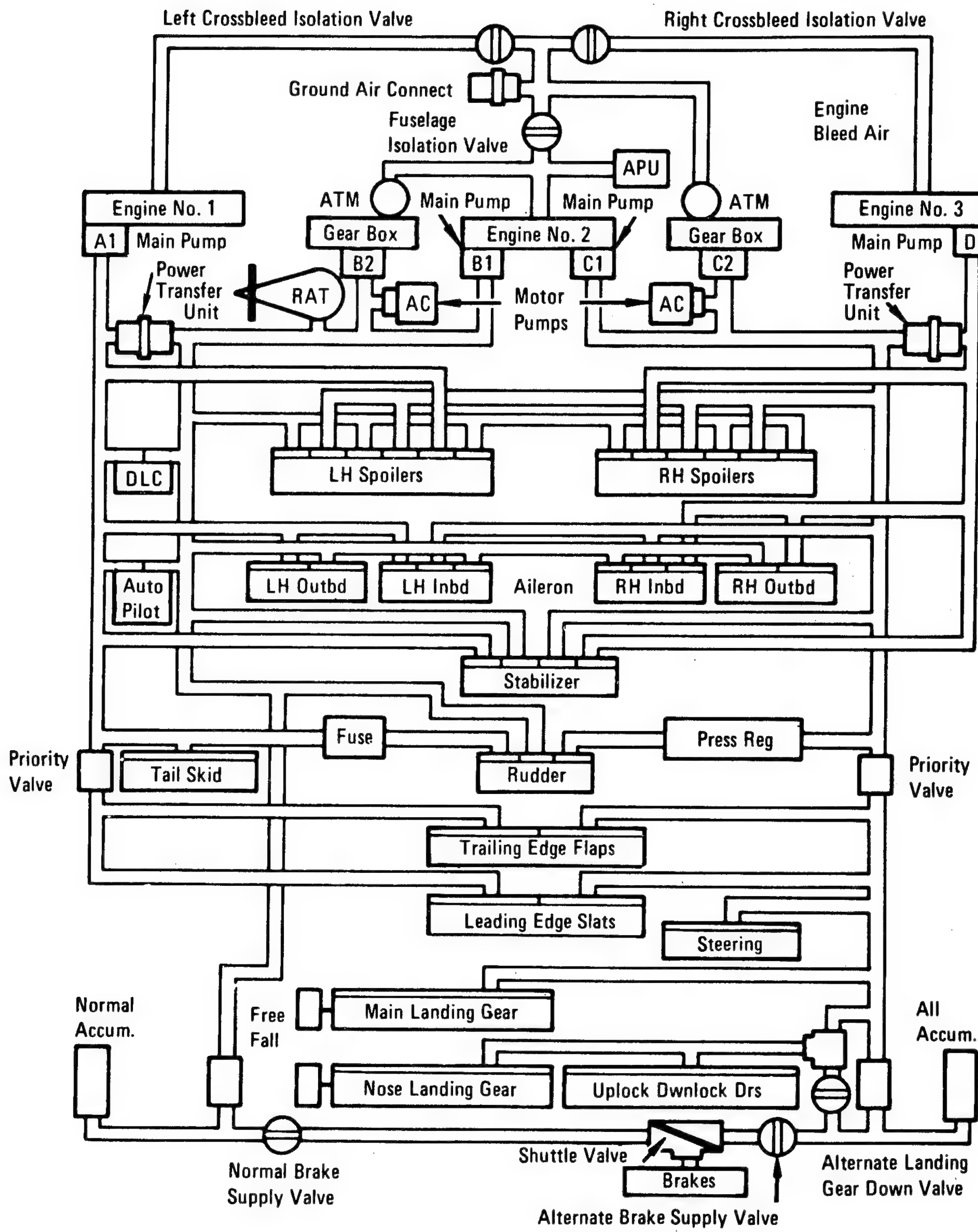


Figure 7. - Baseline hydraulic system schematic.

fluid power system, which furnishes not only quad redundant power channels to the FCS, but also has the back-up support of eight pumping elements and two PTUs. This is the criterion and standard of reliability that must be matched by the electromechanical system, EMAS, for the FCS in the IDEA.

From a power standpoint, the four ED pumps and the two ATM pumps are Vickers PV3-240-2 variable displacement units having a 2.4 in^3/rev displacement each. These pumps deliver approximately 38.8 gpm @ 3880 rpm, so each pump can be rated approximately at 60 hp. The two ac motor pump units are Vickers MTEV3-022-14 having a displacement of 0.22 in^3/rev, having a 10.5 gpm flow capability: the liquid cooled motor, in this unit, is rated at approximately 15.5 hp. Total hydraulic flow demands in the baseline airplane vary significantly, with the system installed capacity being sized by the take-off condition. Figure 8 is a bar chart showing flow condition for each of the systems. Two priority valves are connected into the system downstream of the power take-offs for the FCS, these priority valves isolate all non-essential hydraulic services in emergency conditions.

2.4 ECS

The environmental control system in the baseline airplane follows the design configuration of the system in the Lockheed L-1011-500 aircraft. Basically, the system is powered by engine bleed air that is tapped from the

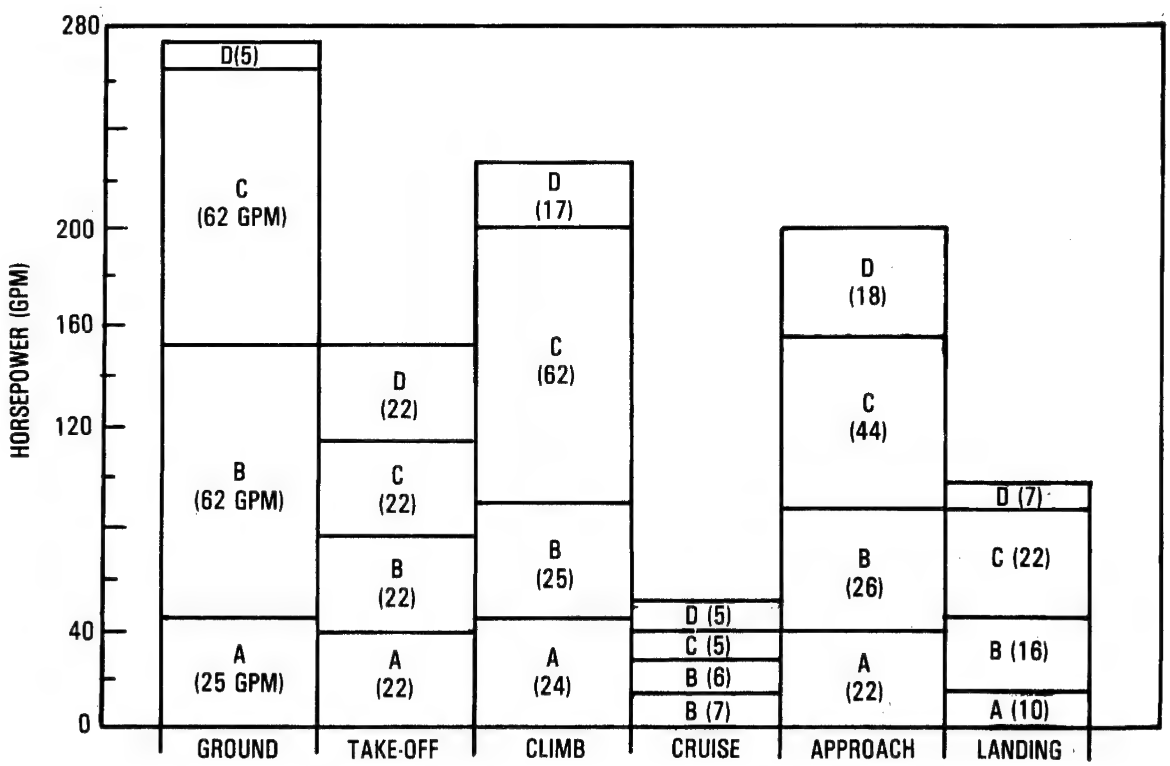


Figure 8. - Hydraulics peak horsepower/system vs. mission segment.

intermediate and high stage bleed of the engine compressors. For reasons of fuel economy, a 50 percent recirculation system is proposed in which 50 percent fresh air is derived by the compressors and 50 percent is fan-recirculated, through filters, from the cabin. The system is designed to provide the following:

a. Cabin pressurization up to 40,000 ft.

b. Heating

c. Cooling

d. Humidification

The pressurization requirement is met by tapping the appropriate pressure stages of each engine compressor, with the equivalent of a fifth stage being used during engine cruise power settings. For low engine power settings and/or the need to supply adequate pressure to operate the cooling turbomachinery during, say, idle descent let-down the high stage bleed would be used. Flow valves regulate the air mass flow from each engine in accordance with the cabin pressure vs. altitude schedule shown in figure 9.

Heating in the baseline is derived from the high temperature bleed air from the engines. Typically, this high temperature bleed air is used for the

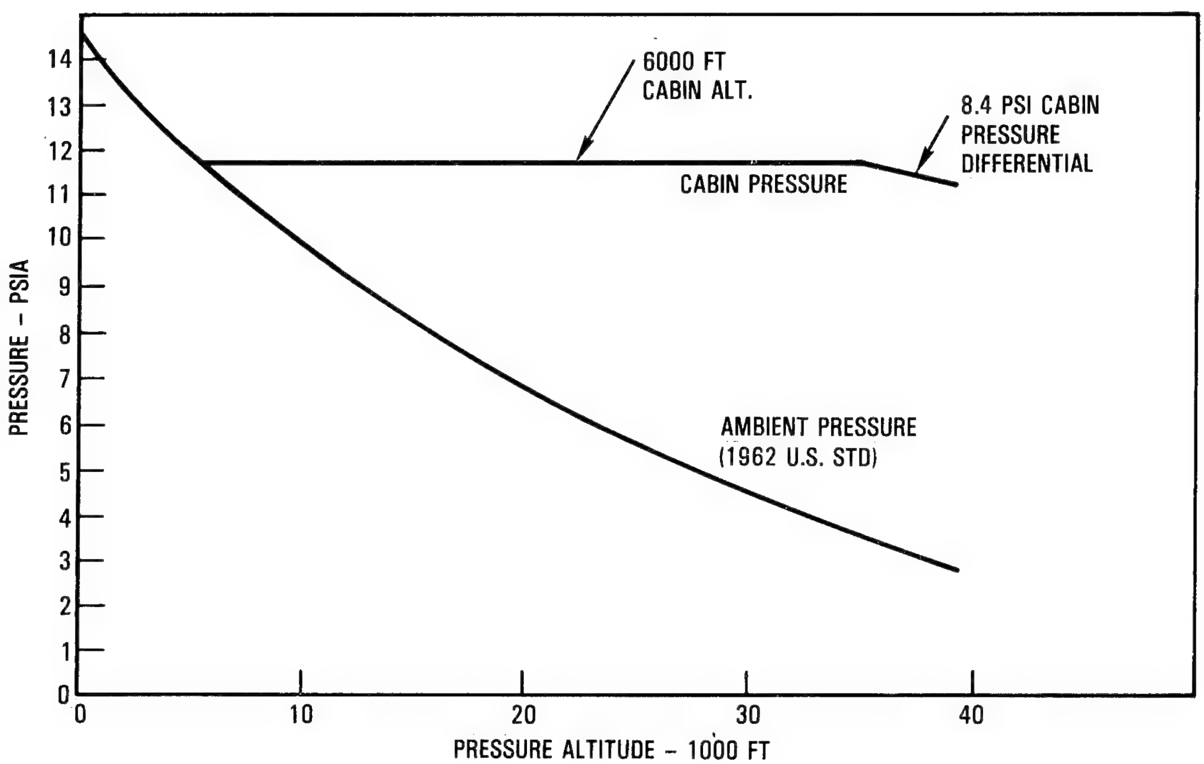


Figure 9. - Cabin pressure schedule.

air conditioning and the ice protection system for the wings and engine: it is also used for powering the ATM driven hydraulic pumps described above. As a consequence, the bleed capacity for high temperature and high pressure air is dictated more by these auxiliary functions, rather than by the ECS, which represents the main bleed power demand on the engines during cruise flight. Therefore, considerable cooling of this air is required, prior to its delivery to the cabin and this is accomplished by: (1) a precooler installation in each power plant; (2) ram air cooled primary and secondary heat exchangers; and (3) air cycle machinery (ACM). Normally, adequate cooling of the bleed air in high altitude cruise flight can be effected with the ram air heat exchangers, permitting the ACM to be bypassed (so as to extend its operational life).

Cooling, as stated above, is accomplished with three types of equipment: a precooler installation in each power plant, primary/secondary (ram air) heat exchangers and expansion cooling machinery. The primary cooling demand appears on the ground with a fully loaded airplane and high outside air temperatures: the burden is also exacerbated by conditions of high humidity. On such days, ground cooling is accomplished with outside conditioned air, or by high-pressure air derived from the APU or any engines (that are running) on the ground. With engines running, air is first cooled to about 500°F by the precooler, which uses bypass fan air as the heat sink; after this, the air is passed through a primary heat exchanger, Hx, and then to the ACM (see figure 10). After compression, the high pressure/high temperature air is again cooled, this time by a secondary Hx, which pulls down the temperature of the air before it is delivered to the expansion cooling turbine. Without external air, or engines running on the ground, the APU will power the ECS via an APU driven load-compressor. With a fully-loaded cabin (350 passengers) on a 103°F day, the cooling system can bring cabin temperature down from 115°F to 75°F in about 30 minutes. Conversely, under cold day operation, with an outside air temperature of -50°F, and a minimal metabolic load, the ECS can maintain the cabin temperature above 70°F.

The baseline ECS bleed schedule is shown in figure 11.

2.5 Flight Controls

The baseline flight control system consists of the primary and secondary flight controls, including stability augmentation, autopilot, and spoilers. The baseline system is similar to the existing L-1011 system but includes active ailerons for gust alleviation, maneuver load control, and elastic mode suppression. The baseline system uses mechanical cable control of servo valves which control full power hydraulic actuators moving the aerodynamic surfaces.

Autopilot and stability augmentation inputs are applied in parallel with the column inputs in the pitch axis and dual mode servo valves in the roll and yaw axis.

Figure 12 is a simplified block diagram showing the electronic flight control system. The automatic flight control computer is digital and

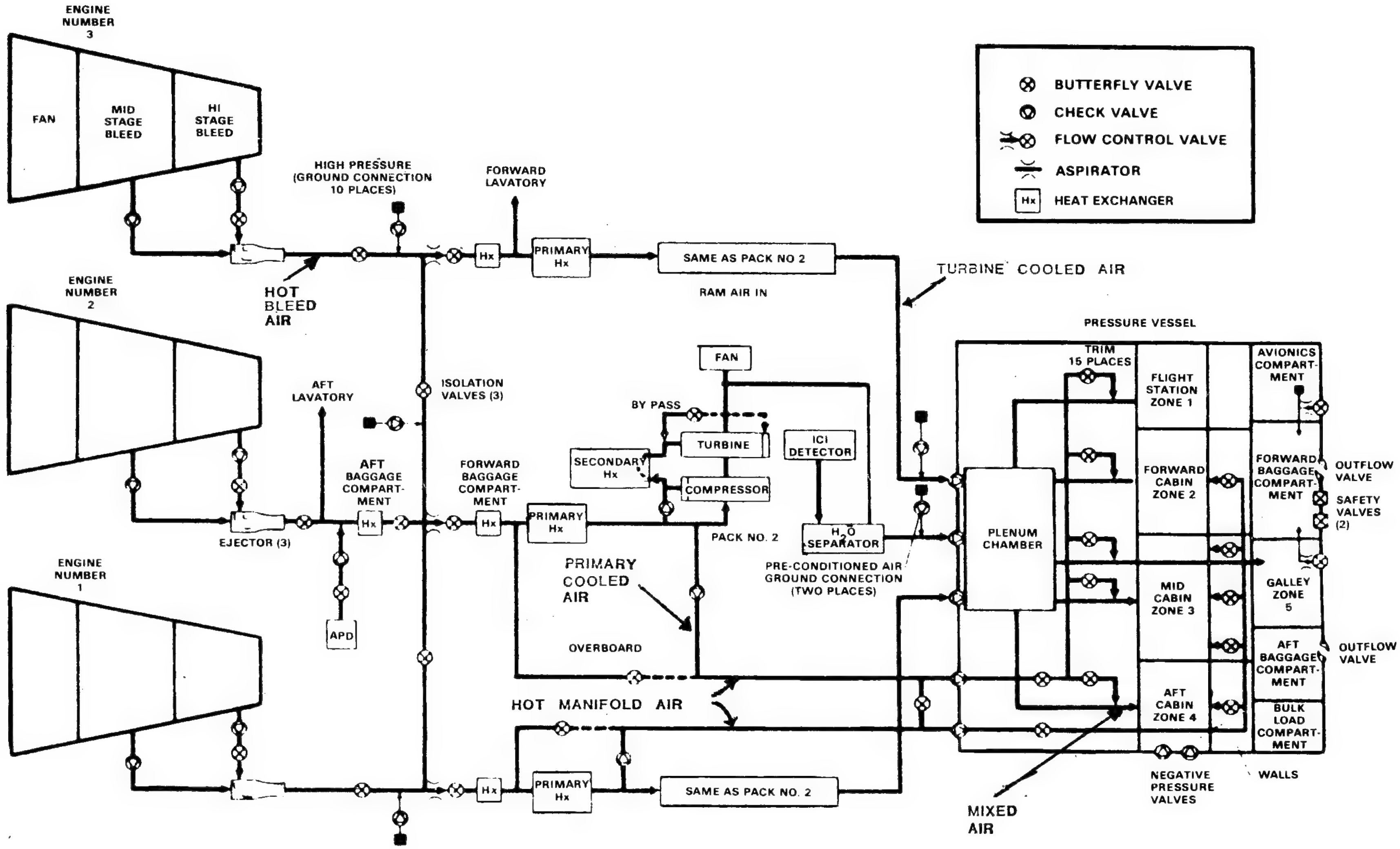


Figure 10. - ECS schematic baseline.

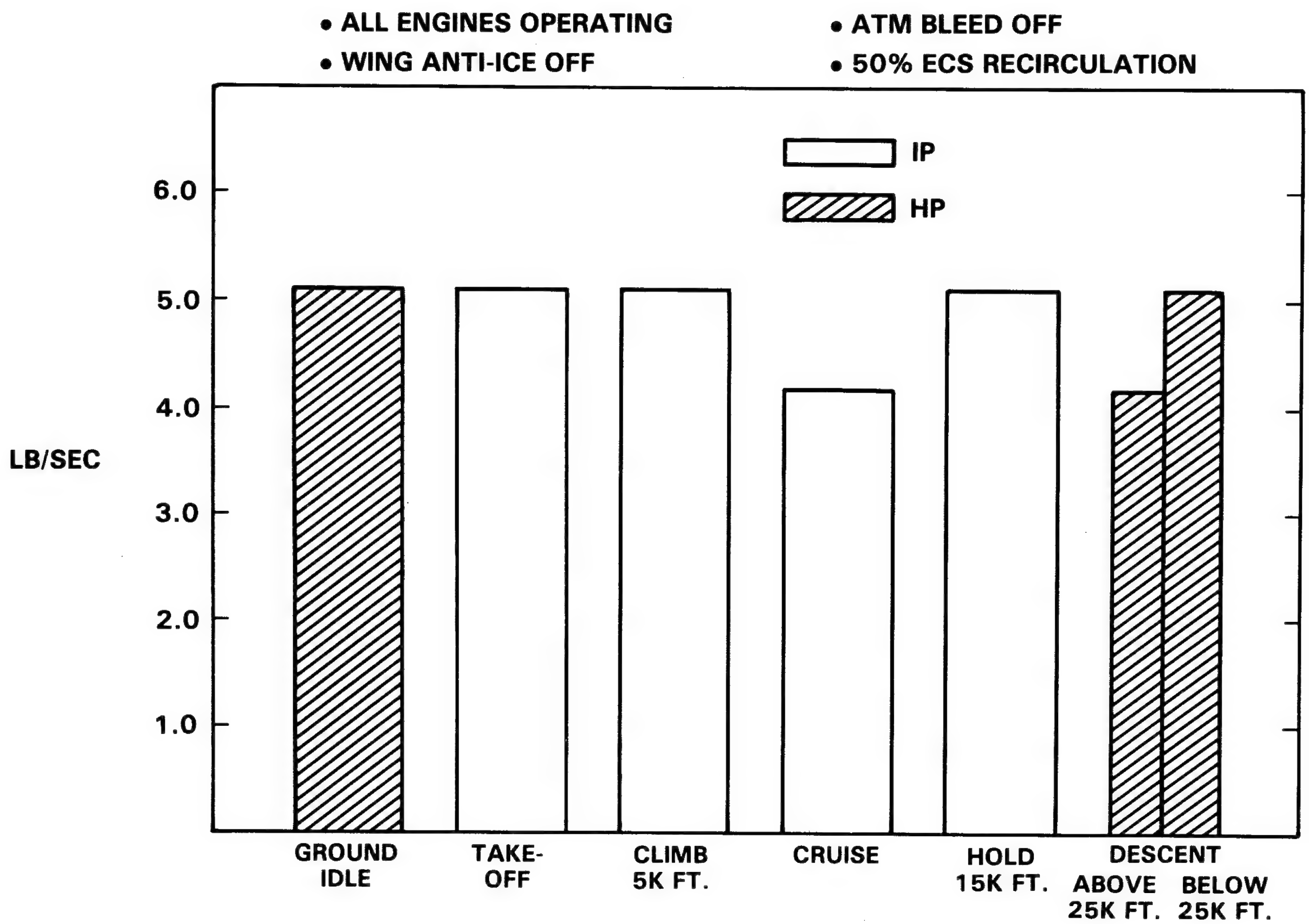


Figure 11. - Bleed requirements for ECS.

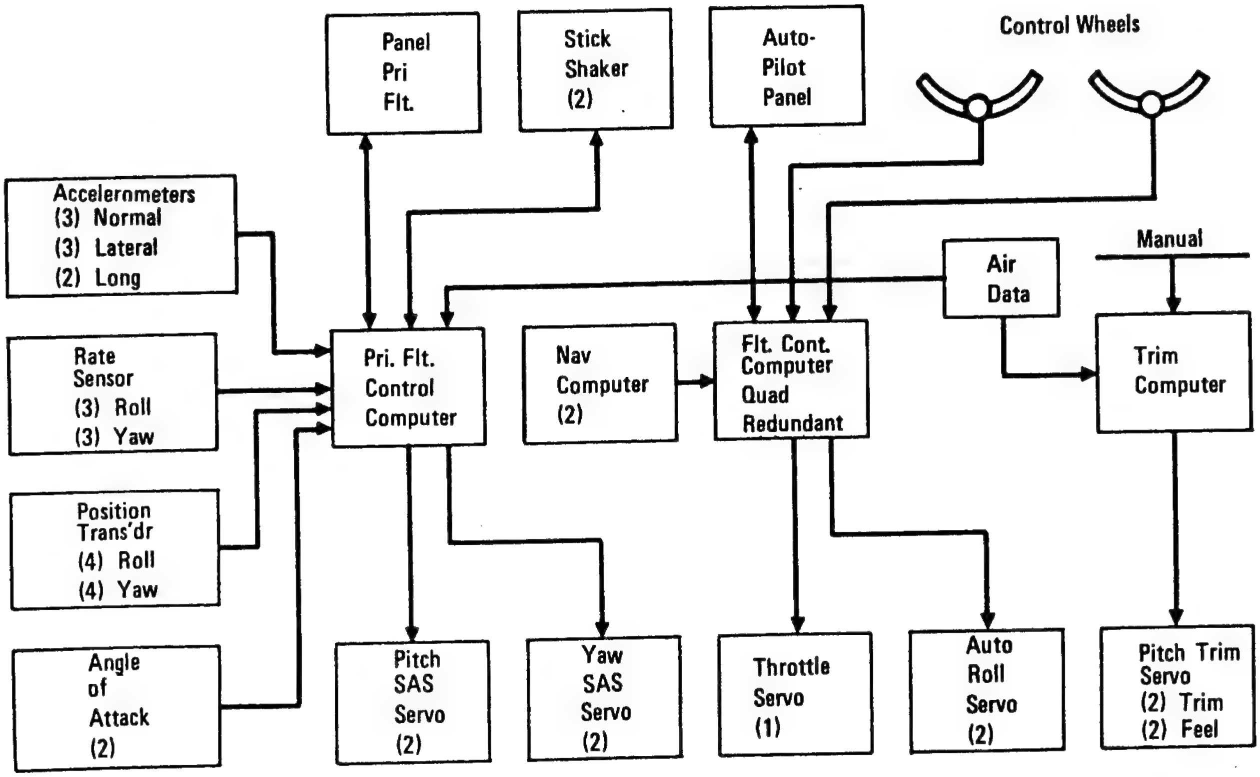


Figure 12. - Baseline digital flight control.

quadruply redundant for the Autoland function. The primary flight control computer is mainly analog and contains stability augmentation circuits, stall warning, altitude alert, system monitor, direct lift control, automatic ground speed brake, and fault isolation monitor. The trim computer provides dual segregated subsystems for manual and automatic pitch trim, Mach trim, and Mach feel. The interconnections to sensors, servos, and instruments are analog; the interconnection with the navigation computer is digital. The significant features of the flight control electronic system are:

- Roll and pitch attitude hold with control wheel steering
- Heading select and hold
- Altitude select and hold
- Vertical speed select and hold
- Indicated airspeed and Mach hold
- Auto control from VOR and area navigation
- Speed control and auto throttle
- Active symmetric aileron control for maneuver load alleviation and gust alleviation

- Cat III ILS and autothrottle
- Takeoff and go-around guidance
- Yaw and nose wheel steering for rollout
- Lift compensation during turns
- Failure protection and warning
- Auto fault isolation

2.5.1 Pitch control. - The horizontal stabilizer rotates for pitch control and trim input (figure 13). The elevator portion is geared to the stabilizer through a nonlinear mechanical drive train for added control effectiveness. Four parallel hydraulic actuators operate in unison to drive the stabilizer. The actuators are controlled by four servo valves each supplied by one of four hydraulic systems. The valves are combined in assemblies of two. Each assembly has one mechanical input linkage and two feedback linkages, one for each valve. The input is mechanically connected to the feedback linkages to close the servo loop. The primary control path is entirely mechanical up to the servo valves; however, this control is modified with powered, limited authority, inputs from the autopilot, trim system, and feel system. The mechanical cable/push rod systems are dual, one for the pilot and one for the first officer (copilot). They are coupled so that both work in unison under normal conditions. The forward coupler can be disconnected manually by the pilot or first officer. The aft coupler located as a part of the stabilizer servo system, is electrically disconnected only when both servos on one side are de-energized. Decoupling, either aft or forward, is required only in case of a system jam.

As the stabilizer leading edge moves from one degree up to 14 degrees down, the geared elevator moves in the same direction as the stabilizer from zero (faired) to 28 degrees trailing edge up.

- Pitch Feel and Trim System: The trim motor, operated by a manual switch on the control column, is primarily a combined series/parallel trim to decrease column excursion required for trimming. The pilot's feel force is the product of control column displacement from trim and the feel spring constant. The trim motor is also controlled automatically by the autopilot when engaged, and by the Mach number to compensate for movement of aerodynamic center of pressure.

  The pilot may override the output of the trim motor with a manual trim wheel through cable, gears, and a ball clutch. The feel force is a maximum of 85 pounds at the column and can be overriden by the pilot. No matter where the trim is set, the pilot can obtain full excursions of the stabilizer with reasonable column forces.

- Pitch Monitoring System: A monitoring system detects jams and open links in the mechanical system. The sensing system consists of bungees (springs) in the cable systems and aft coupler that are instrumented to

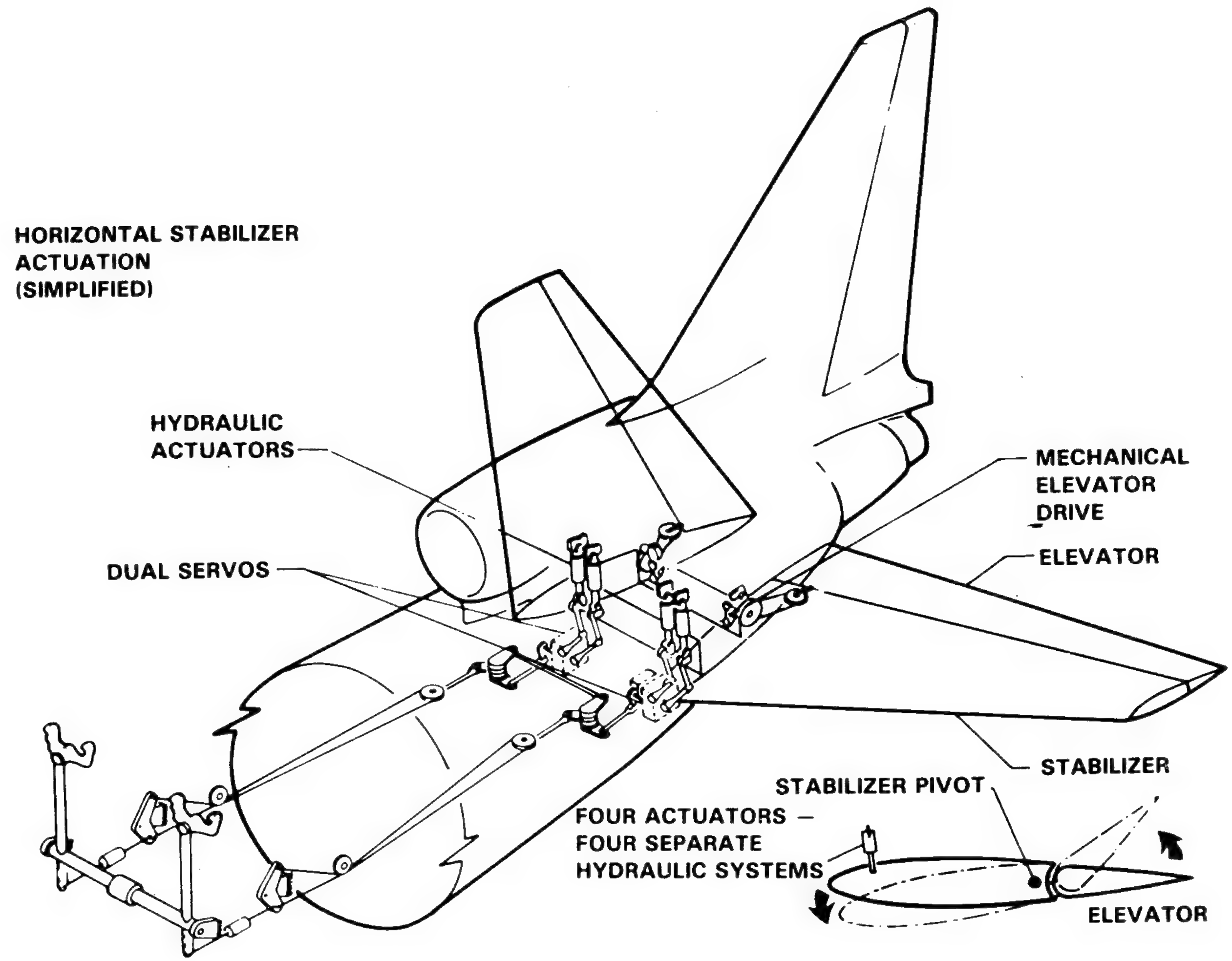


Figure 13. - Baseline primary FCS: horizontal stabilizer.

detect motion when the force exceeds bungee preload force, and cable integrity sensors instrumented to detect loss of continuity. A logic network uses the signals to determine the location of the jam or open and the appropriate action required. Warning lights direct the pilot to remove hydraulic power from the appropriate servos and manually disconnect the forward coupler. The aft coupler open automatically when power is removed from the servo valves. Control is maintained by the redundant cable system and the remaining set of servos, however, the feel force is reduced to one-half of normal when the coupler is open.

- Stall Warning System: An artificial stall warning is provided by means of two shakers which vibrate the pilots' control columns whenever the aircraft speed is less then 1.07 times the stall speed. The stall speed is computed using a combination of air data, angle of attack, slat, and flap positions. The system is inoperative when the landing gear struts are compressed (aircraft is on the ground). The system commands the spoilers to retract when a stall warning is indicated. Sensor and power faults are annunciated in the cockpit, and channel selection capability is provided.

2.5.2 Roll control system. - Pilot control inputs are communicated mechanically from the control wheels to the servo valves at the ailerons (figure 14). Separate paths are provided from each control wheel to the inboard aileron on the corresponding side (left or right). In normal operation the control wheels are coupled and the left and right ailerons operate in unison asymmetrically. If a jam occurs, the wheels can be manually decoupled.

All four aileron surfaces deflect $\pm$ 20 degrees. Aileron roll control is supplemented by spoilers during low speed (flaps extended) flight. Spoiler deflection is a nonlinear function of aileron deflection with 40 degrees of up spoiler corresponding to 20 degress of up aileron on the same wing. Similarly, 2.5, 12.5, and 17 degrees of aileron correspond to 0, 10, 20 degrees of spoiler, respectively.

- Aileron Servos: Three hydraulic actuators and three servo valves serve each inboard aileron; and two actuators and two servo valves serve each outboard aileron. Each actuator for a particular aileron is supplied by a separate hydraulic system. The servo valves for a particular aileron are assembled with a common input torque shaft. Two feedback rods are provided at each servo valve. Two input rods are provided at the inboard servo valves, one at the outboard. The dual input and feedback rods operate on opposite ends of the common input torque shaft for the servo valve assembly. In addition to mechanical commands, two of the three left inboard servo valves accept electrical commands from the autopilot. When on autopilot, the position of the left inboard aileron is fed mechanically to the other ailerons through the primary mechanical system.

- Roll Feel and Trim: Artificial feel and centering for the roll control system is provided by a single compression spring cartridge in the left control path. The ground point of the feel spring is shifted by the

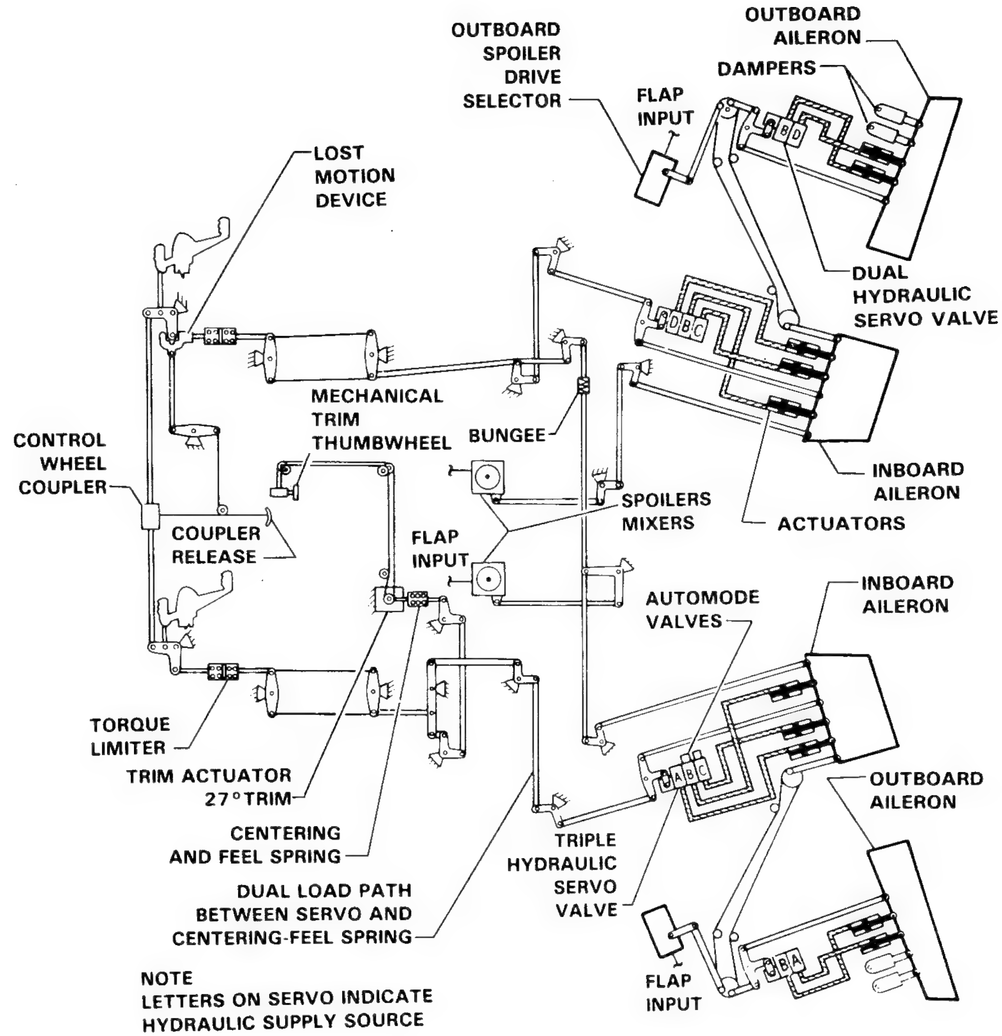


Figure 14. - Baseline primary FCS: roll axis - aileron.

roll trim actuator, thereby providing parallel roll trim. Overtravel is provided so that full roll control is available irrespective of the trim actuator position. The trim system can provide up to $\pm$ 7 degrees of aileron travel. Spoiler operation is affected by aileron trim in the same manner as by other aileron inputs.

- Monitoring System: Two torque limiters and a cross-tie bungee are included to permit continued roll operation in the event of opens or jams in the mechanical control paths. The cross-tie bungee does not have a deflection switch but it does permit relative motion between the two ailerons. The torque limiters each permit relative motion between control wheels and cable system and contain sensors to detect deflection for use in the monitor display system. If a jam occurs downstream of the limiter in either control path, continued control is possible by overcoming the breakout force of the affected limiter and controlling through the other control patch. Operation of the torque limiters is displayed to the pilot for manual shutdown of the affected aileron and spoiler actuators.

The modulating signal for direct lift comes from the autotrim transducer in the autopilot pitch servo. It does not depend upon selection or engagement of the autopilot and is essentially a stabilizer-out-of-trim signal. Altitude changes are thus produced largely from operation of the DLC spoilers rather than the stabilizer, with much reduced pitch attitude excursions.

Spoiler automatic operation for landing, rejected takeoff, go-around, and incipient stall is determined by logic in the flight control electronic system. Inputs are from flap handle, throttle levers, thrust reverser levers, stabilizer control system, landing gear control handle and landing gear strut compression. During a normal landing, when landing gear is down, flaps are extended, and landing gear switches indicate aircraft touch-down, the computer asks for 12 degrees spoiler deflection after a half-second delay, then when struts are fully compressed, spoilers extend to 60 degrees. Limiting rudder deflections is accomplished by dual positive mechanical stops operated by solenoid operated hydraulic actuators. There are four rudder actuators arranged in two dual tandem sets. Three servo valves are provided assembled side by side with separate input push rods to each side of the common input shaft. Each servo valve has input from a separate hydraulic system (A, B, and C). One valve serves two actuators. Two of the valves have electrical inputs in addition to the mechanical input. The electrical input is used for yaw stability augmentation.

The rudder is controlled automatically for dutch roll damping and turn coordination during all phases of flight and for runway alignment and roll out during autoland. In the basic stability augmentation system (SAS), the control is independent of autopilot status and allows pilot inputs to be added via the rudder pedals. SAS and turn coordination are achieved by processing inputs from the three rate gyros and four aileron position transducers. For approach and land, the aileron signals are switched out. The runway alignment signal is a function of instrument landing system (ILS) error, heading error, altitude, and yaw rate. The alignment scheme is a limited forward slip maneuver in which up to eight degrees of initial crab angle is removed by

lowering a wing and slipping the aircraft. After touchdown, the autoland computation uses ILS error and yaw rate to direct the aircraft down the runway with rudder control and limited nose wheel steering.

2.5.3 Autopilot. - There are four channels in each axis for approach and landing, and there are only two which are active for cruise. The system has two dual computers, autopilot A and B which can be engaged independently or simultaneously, either in the autopilot mode (in approach/land only) or flight director mode. Thus, either or both flight directors may be used to provide flight director steering information to the pilot, with or without autopilot engagement. With autopilot engagement the flight director may be used to monitor autopilot operation. Each pitch system (A and B) has a servo with mechanical input into the mechanical control. The roll output (A and B) is electrical, direct to the aileron actuator servo valves of the left inboard aileron. In either case, the autopilot outputs operate in parallel with the control wheel inputs. The pilot can mechanically overpower the autopilot servos through the control wheel.

The basic autopilot mode is parameter hold with the pilot able to input change through control wheel steering. The autopilot command mode provides automatic control in response to a computed guidance signal.

An automatic trim system acts to center the autopilot servos to prevent transients when the autopilot is either manually or automatically disengaged. There are two automatic pitch trim systems and at least one must be operative to engage either autopilot. The altitude signal for altitude hold and altitude select is a rate-and-displacement-limited barometric altitude error signal which is gain scheduled as a function of true airspeed. An integration path is provided to compensate for long term error signals. The control signal is mixed with pitch attitude and attitude rate signals for control loop damping. As the altitude approaches the selected altitude, the altitude rate and altitude error are used to compute the point at which the maneuver to capture the desired altitude is initiated. At initiation, an exponential flare maneuver to capture the desired altitude is commanded. When the maneuver is completed, the altitude hold mode is automatically established and annunciated.

Roll attitude/heading hold is the basic roll axis autopilot mode. Upon engagement, the autopilot will maintain heading if the bank angle is less than five degrees and will maintain bank angle if over five degrees. Control wheel steering can be used to establish a new roll attitude or heading reference.

In the navigation mode, the autopilot will direct the aircraft to capture and follow a VOR beam or an Area Nav course, if these systems are operating.

The approach/land mode will capture the localizer beam, follow the localizer beam, capture the glideslope, follow the glideslope, align with runway at 150 ft altitude, perform flare at 50 ft altitude, and maintain heading down the runway on rollout.

The glideslope capture maneuver is inhibited until localizer track is established and glideslope deviation is less than 30 microamperes. The flare

gain is scheduled as a function of radio altitude, radio altitude rate and normal acceleration to essentially zero rate at zero altitude.

The turbulence mode is normally engaged when the aircraft is flying in turbulence. The autopilot reverts to the parameter hold configuration with reduced gains to provide softer control.

2.5.4 Stability and control. - The horizontal tail is sized by the requirement for a total c.g. travel of 4.8 ft with 6 percent mean aerodynamic chord (MAC) static margin at the aft c.g. limit. This results in a horizontal tail volume coefficient of 1.08 and c.g. limits of 17 percent to 41 percent MAC (figure 15).

2.6 Aerodynamics

Airfoil characteristics are based on trends in improvements in airfoil design (such as drag divergence Mach number). These characteristics are incorporated into a parametric aerodynamic performance program which predicts drag polars for wings of different sweep, aspect ratio, thickness/chord ratio, etc. A description of this method may be found in reference 3.

2.7 Propulsion

The General Electric Energy Efficient Engine (E^3) was used for the baseline aircraft and IDEA performance evaluation. The engine performance was

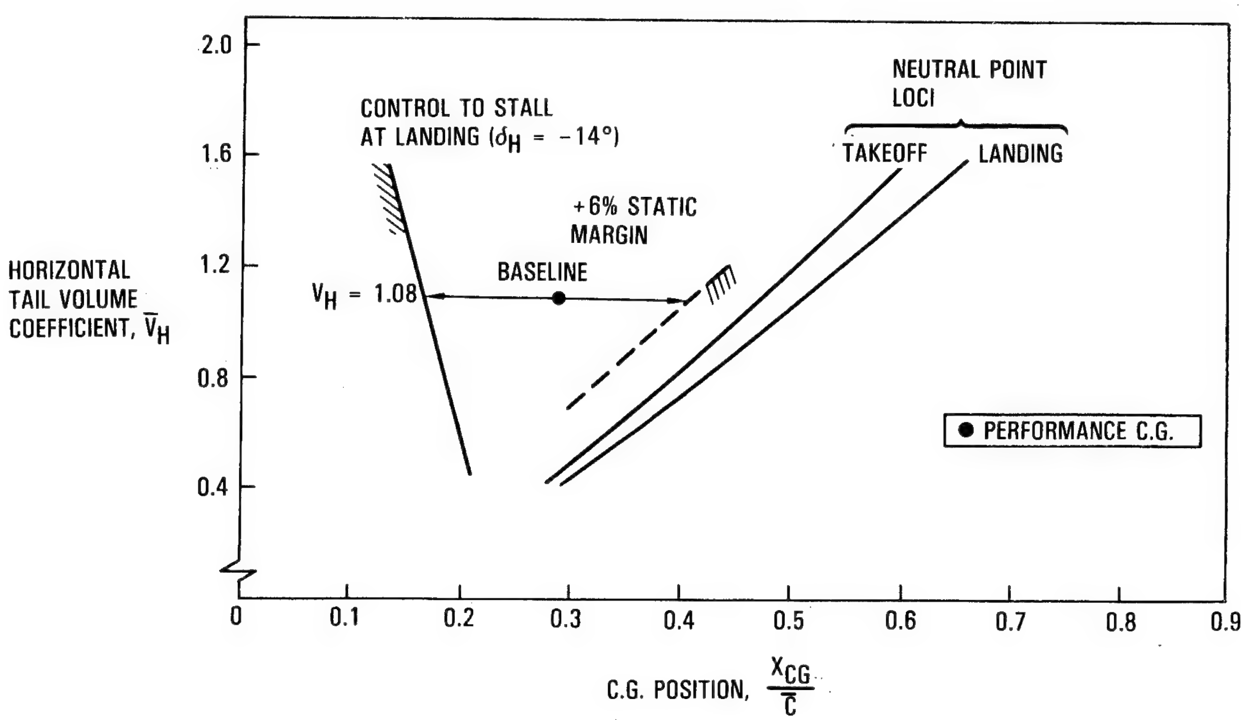


Figure 15. - Horizontal tail sizing.

estimated by GE for a fully developed propulsion system based on the results of the recently completed NASA funded E^3 program. The engine technology required to provide the estimated performance would be available in the mid 1990s.

The engine gearbox for the aircraft accessories is core mounted and driven off the high pressure compressor shaft. For the baseline aircraft, the aircraft accessories include generators, hydraulic pumps, engine starter, plus the engine required accessories (fuel and oil pumps). High pressure compressor bleed air is used for the environmental control system on the baseline aircraft. Power requirements are defined in Sections 2.2 through 2.4.

The General Electric E^3 engine/nacelle configuration is illustrated in figure 16. This configuration uses a mixed flow exhaust system. The propulsion system characteristics are given in table 2. The engine bypass ratio and overall pressure ratio are higher than current operational engines, however the turbine inlet temperature has not been significantly changed. Specific fuel consumption has been improved approximately 15 percent relative to current high bypass engines.

2.8 Structures and Materials

For each configuration aggressive use is made of graphite/epoxy composite materials. Advanced aluminum alloys are used where they are more effective

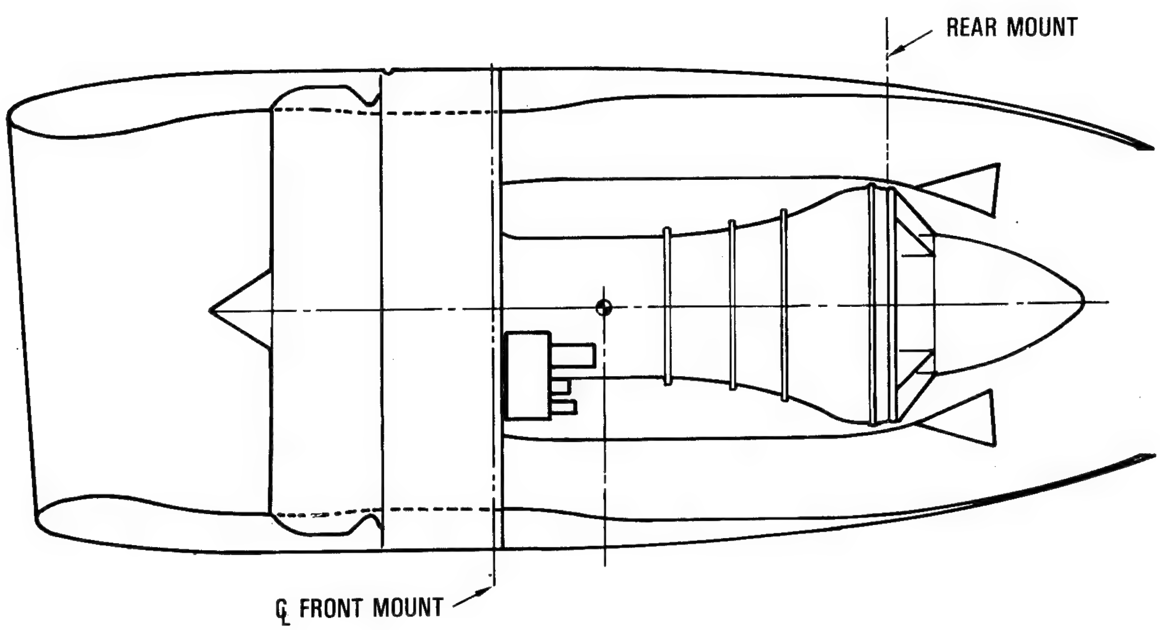


Figure 16. - NASA/GE E^3 FPS-9 engine/nacelle.

TABLE 2 - BASELINE PROPULSION SYSTEM CHARACTERISTICS.

	Aluminum	Titanium	Steel	Composite	Other	% Weight Reduction
Wing	0.11	0.04	0.05	0.77	0.03	0.21
Tail	0.10	0.10	0.06	0.59	0.15	0.21
Fuselage	0.44	0.07	0.01	0.40	0.08	0.11
Landing Gear	0.20	0.00	0.33	0.00	0.47	0.05
Nacelle	0.11	0.57	0.13	0.18	0.01	0.00
Engine Section	1.00	0.00	0.00	0.00	0.00	0.02
Air Induction	0.73	0.04	0.00	0.23	0.00	0.09

than composites. The distribution of material and percentage weight reduction (as compared with a conventional aluminum structure) are shown in table 3.

2.9 Baseline Optimization

2.9.1 Sweep, aspect ratio, and thickness/chord ratio. - Wing sweep, aspect ratio and thickness/chord ratio are based on the advanced configuration described in reference 4. At the time the study described in reference 4 was performed fuel prices were rising in constant dollars, and it appeared that minimization of block fuel was of paramount concern in aircraft design. The wing was therefore designed to minimize block fuel (figure 17). Trends in fuel prices (in constant dollars) are now stable or declining, that wing design to minimize DOC is more appropriate. Because of the design constraint in fitting the landing gear, aspect ratio was limited to a maximum value of 12. The resulting wing design is close to that for minimum DOC using the price ground rules of this study. This wing design is therefore considered suitable for the Baseline.

2.9.2 Thrust/weight ratio and wing loading. - A carpet plot of thrust/weight ratio (T/W) and wing loading (W/S) is shown in figure 18. The constraint that the aircraft should have 180 ft/min rate of climb at the top of climb, using maximum climb power, requires that T/W have a value of about 0.25. The fuel volume constraint (which requires the wing fuel volume to be 10 percent greater than the mission requirements) constrains wing loading to be less than 131 lb/ft. These two constraints size the baseline aircraft.

2.10 Configuration Description

The following sections summarize some of the leading characteristics of the baseline configuration. A detailed description may be found in Appendix A.

TABLE 3. - MATERIAL DISTRIBUTION AND WEIGHT REDUCTION

	Aluminum	Titanium	Steel	Composite	Other	% Weight Reduction
Wing	0.11	0.04	0.05	0.77	0.03	21
Tail	0.10	0.10	0.06	0.59	0.15	21
Fuselage	0.44	0.07	0.01	0.40	0.08	11
Landing Gear	0.20	0.00	0.33	0.00	0.47	05
Nacelle	0.11	0.57	0.13	0.18	0.01	00
Engine Section	1.00	0.00	0.00	0.00	0.00	02
Air Induction	0.73	0.04	0.00	0.23	0.00	09

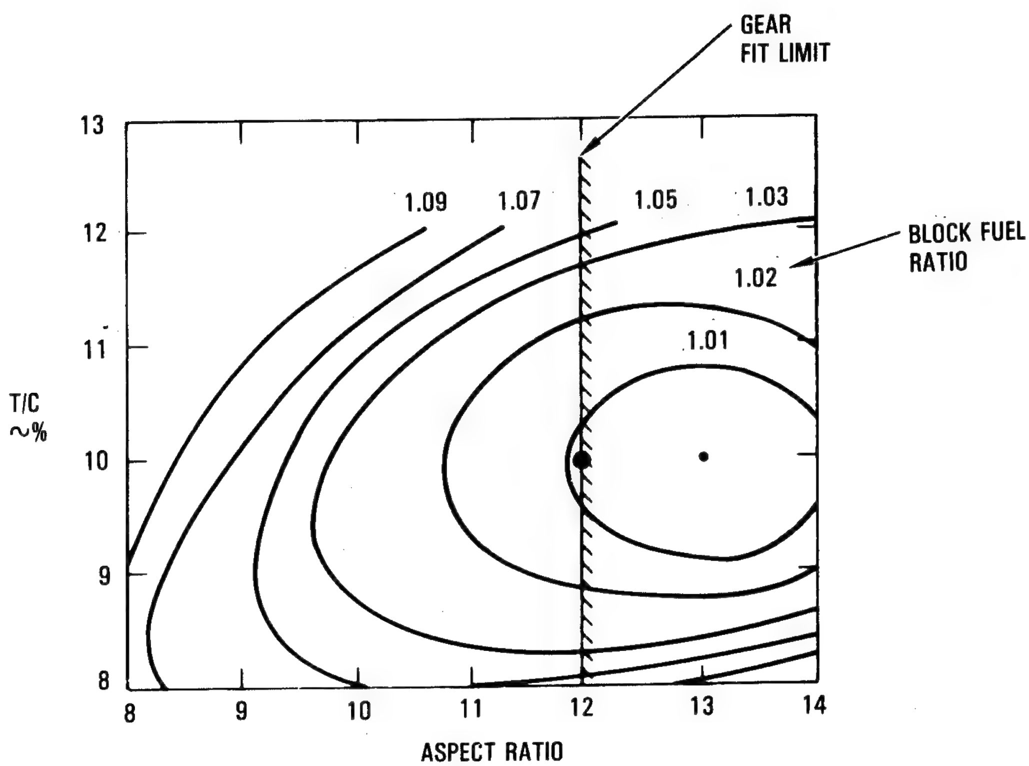


Figure 17. - Wing parameter optimization.

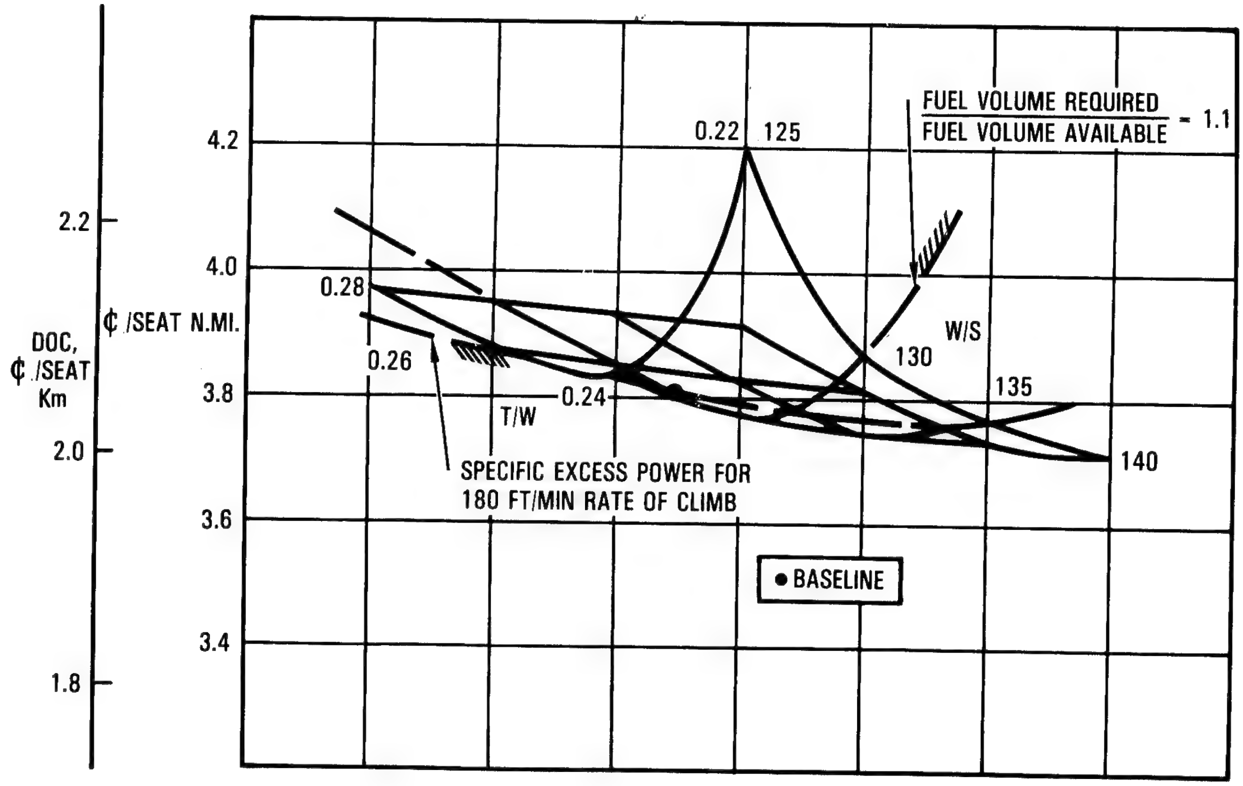


Figure 18. - Direct operating cost at 2500 n.mi.

2.10.1 Geometric description. - Table 4 summarizes leading geometric data. A three-view of the baseline configuration is shown in figure 19.

2.10.2 Weights. - Baseline aircraft weights are summarized in table 5.

3. ADVANCED POWER GENERATION SYSTEMS

This section discusses the comparison of variable speed constant frequency (VSCF), constant speed drives (CSDs), variable voltage/variable frequency (VV/VF) and high frequency power-generation systems. The guidelines that dictate this comparison are, to a major extent, related to their suitability to the IDEA and Alternate IDEA airplanes.

3.1 Historical Background

The aerospace electrical industry has pursued the development of power generation systems which would best meet the needs of the utilization equipment. Pre-World War II, the electric/avionic power demands were met by 12 V, and later by 24 Vdc systems, since the ac power requirements at the time were only 500 to 1500 W. This power was used mainly for lighting, communication-radios, and minimal armament. Towards the end of World War II, however, the

TABLE 4. - BASELINE CONFIGURATION GEOMETRY

Wing	Trapezoidal Area	3590 ft^2
	Span	207.5 ft
	Taper Ratio	0.246
	Sweep at 1/4 chord	25°
	MAC	19.4 ft
Horizontal Tail	Trapezoidal Area	955 ft^2
	Tail Vol. Coeff	1.08
Vertical Tail	Trapezoidal Area	567 ft^2
	Tail Vol Coeff	0.059
Fuselage	Length	201 ft
	Diameter	19.6 ft

TABLE 5. - BASELINE AIRCRAFT WEIGHTS

	Weight, lb	Weight Fraction, %
Gross Weight	470224	
Fuel Available	142908	30.39
Zero Fuel Weight	327316	
Payload	73500	15.63
OEW	353816	
Empty Weight	233940	
Structure	53785	30.50
Propulsion	26004	5.53
Systems	64527	13.72

electric power demands had grown to the point that four 28 V 300 amp (9 kW) generators were required to support the electrical needs of the larger bomber aircraft and the emerging large commercial transports. The avionics ac load requirements in these aircraft were then met by inverters (consisting of dc motor-driven/ac generator sets) that ranged in size from 100 VA up to 2500 VA each. The specific weight of the earlier dc generators (and dc starter-generators) averaged 43 lb/kW, mainly because of the wider speed range of the piston engines and the lower base-speeds of the generators.

In the early and late 1940s, the U.S. and British Advisory Staffs, recognizing the trend towards the much larger capacity power generation systems, advocated the use of primary ac generator systems with static rectifier units to furnish 28 Vdc power for the large amount of 28 Vdc utilization equipment. Nonetheless, the 28 Vdc power requirement was 'secondary' in relation to the projected capacity of the new ac (primary) power systems. In addition to the selection of a suitable higher transmission voltage for the ac generators, there were controversial discussions as to the choice of an optimum frequency for these new power systems. The industrial frequencies of 50 and 60 Hz were

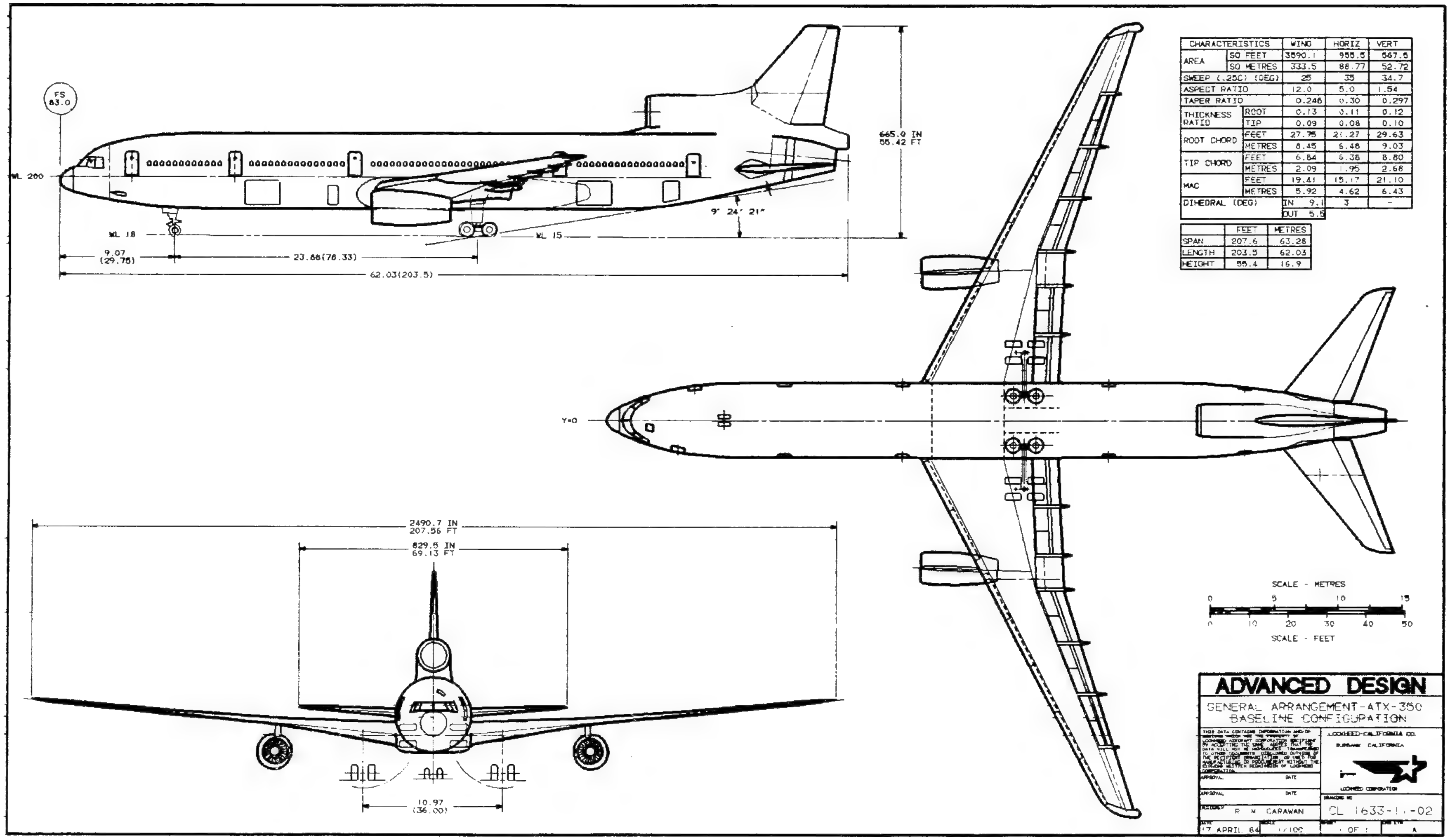


Figure 19. - Three-view baseline configuration.

recognized as being too low for the aerospace industry, so frequencies of 250, 400, 600, 800 Hz (and higher) were evaluated. Finally, the 200 V/400 Hz power was selected as a standard, since it permitted ac motors to run up to speeds of 24,000 rpm and it offered attractive reductions in the electromagnet weight for motors, transformers, filters, etc.

Figure 20 is a generalized plot of the motors/electromagnet weight saving versus frequency. This shows that substantial reductions were made relative to 50/60 Hz equipment, but above 400 Hz, increasing iron losses, problems of skin effect, hysteresis, eddy-current losses, etc., resulted in decreasing weight gains. With the development of low loss irons (Metglass, etc.) and the trend to high speed rotating machinery, it is reasonable to consider 600, 800 Hz primary and even higher frequency ac systems.

The developments in aircraft engine technology have had a significant impact on aircraft power systems. The new turbojet and turbofan engines have narrower speed ranges, typically 2:1 compared to 3:1 or 4:1 for the earlier piston engines. This impacted favorably on the weight of the new synchronous generators since the specific weights of ac generators dropped to as low as 1.2 lb/kVA and, today, to less thann 1.0 lb/kVA, (for modern high-speed ac generators).

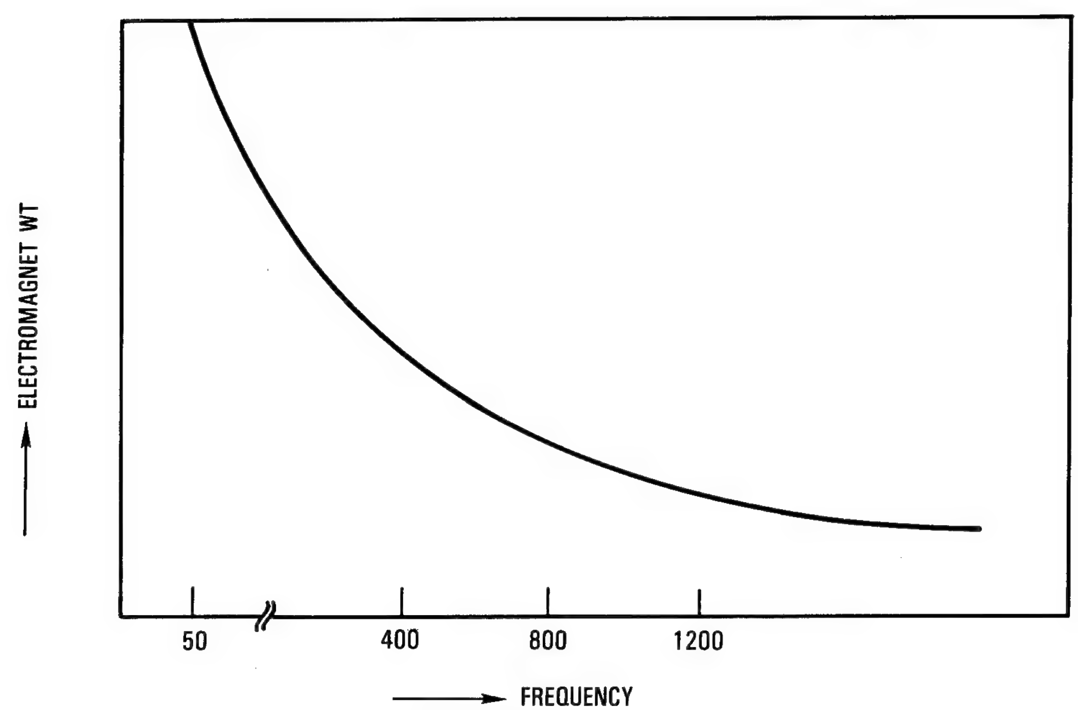


Figure 20. - Motors/transformer weights vs. frequency.

3.1.1 Engine speed impact. - The power generator system performance and weight are therefore closely related to the engine in terms of its speed range and the speeds of the accessory pads. For the conventional 400 Hz synchronous generators, Military Standard (AND 20006) pads having nominal speeds of 6000 and 8000 rpm were provided. For the synchronous generators, such as the VSCF generators, however, the speed requirements were different since a cycloconverter VSCF system requires a 4:1 (input-frequency to output-frequency) ratio and therefore very high generator speeds. The dc link type VSCF systems has no special frequency constraint, but still requires high generator speeds since it is in the interests of both systems to utilize high speeds. Thus, the VSCF generator in both systems each benefit from high speeds in the range of 10,000 to 20,000 rpm.

Where direct driven generators are used and there is no major in-line power conversion, the generators are running at approximately constant speed over the major part of the operating flight range of the aircraft engines. Therefore, with the right selection of the number of generator poles and the right speed, hybrid type systems of the variable voltage/variable frequency type can be designed to operate at nominal frequencies of 400, 600, 800 Hz, at say the 92% cruise speed of the engines.

3.2 Power System Alternatives

The engine speed range, and the actual engine speeds from ground-idle to take-off thrust rating, are therefore critical to the performance of the power generator system. Typically, however, in most current ac electrical systems the ac generator is driven by constant speed drives (CSDs), so the generator designs are optimized via the technology resident in CSDs. Because of the generator's constant speed operation, the generators are able to furnish the same power at ground idle speed as they are at takeoff engine speed. The same is true of the early dc systems but, in this case, it was necessary only to keep the voltage constant, whereas in the ac systems it was more complicated because the voltage and the frequency had to be held constant. It can therefore be seen that the CSDs provided aircraft generator with the luxury of constant speed operation, but a legacy of this was that the cost and complexity of the power generating channels increased. More importantly, the system reliability system overhaul time reduced to the disadvantage of the life cycle costs. Partly as a result of this, electric/electronic approaches of the VSCF type systems were sponsored and developed under Military and industry funding. Also in the recent NASA/Lockheed studies (ref. 2) a variable voltage/variable frequency system (VV/VF) which had a "power-proportional-to-speed" characteristic was proposed, because few aircraft require the same amount of power at engine ground idle, as they do at takeoff (or cruise). One concern for the VV/VF systems was how large motors and transformers would operate from such a power supply. To address this, figure 21 shows the performance of the ac induction motors operating at different frequencies, when (1) the voltage is held constant and (2) when it is allowed to change with speed (frequency). Note that where the voltage is held constant, the motor develops excessive torque at the lower frequency and, normally, it cannot be absorbed by most loads (such as compressors). In contrast when the V/F ratio

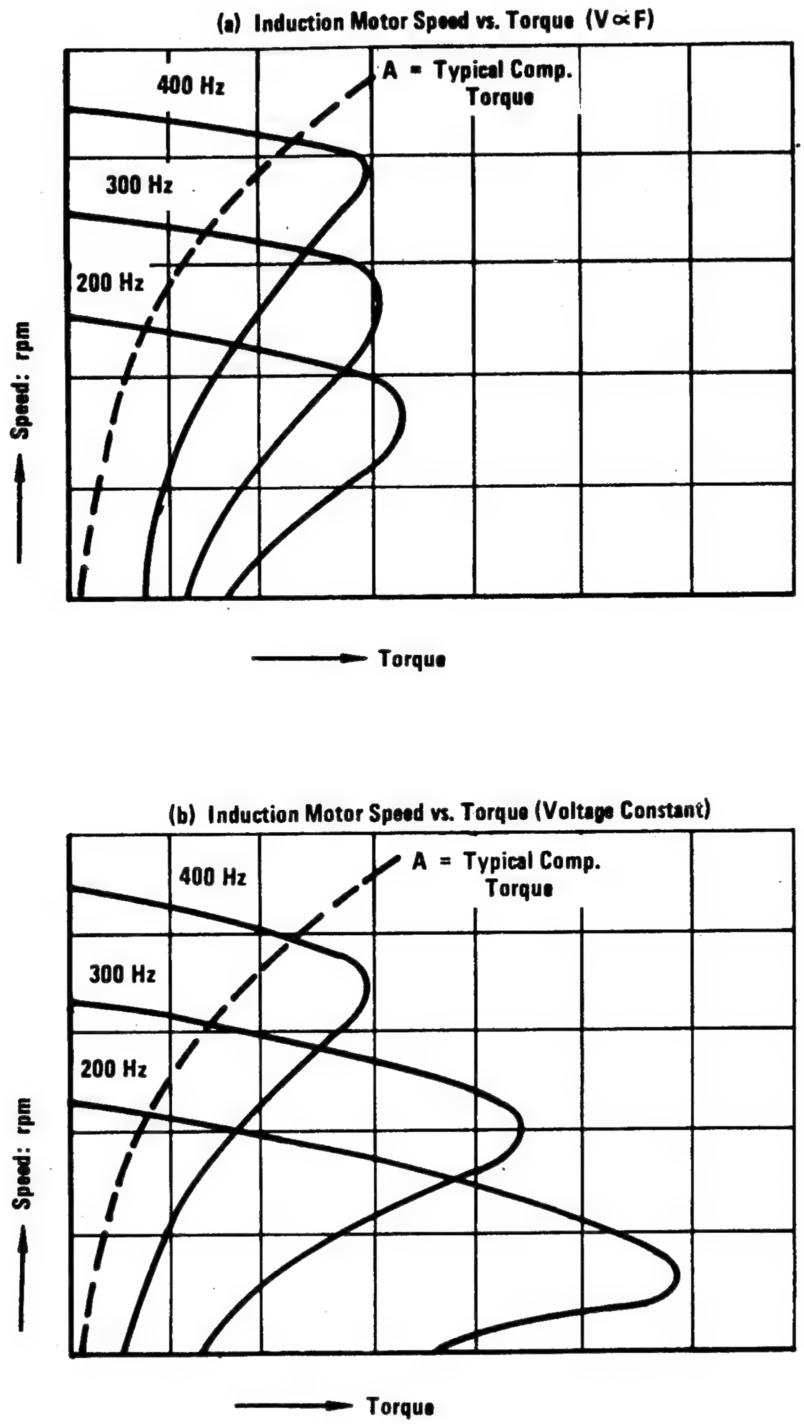


Figure 21. - Induction motors: torque/speed characteristics.

is held constant, the motor stator-flux, (and the motor torque) remain constant over the speed range so this is very satisfactory for most loads in the aircraft.

The development by the U.S. Air Force and Navy of systems using electronically derived methods of providing constant frequency from variable frequency power sources has been looked upon as alternative methods of providing high-reliability/high-quality electric power, since it was thought that with no moving parts, the life cycle costs and reliability aspects could be improved. Also it was claimed that a direct engine driven generator would provide improved overall transmission efficiency in the power plant and therefore reduce the heat rejection losses in the engine nacelle area.

More recently, NASA-Lewis has been examining the benefits and potential of a 20 kilohertz power transmission, which could take advantage of bidirectional resonant frequency switching mode inverters/converters that would permit conversion of this high frequency power to other forms of power, typically required in a modern aircraft. Also, because of the high frequency, NASA-Lewis should look to the use of Metglass and advanced materials/wire developments to take advantage of this type of high frequency transmission system.

There are many possible aircraft power generating systems, and each has varying degrees of merit; some also are limited because of their development and lack of implementation status. The following delineates the systems to be analyzed.

1. Constant Speed Constant Frequency, CSCF (such as the present CSD systems).

2. Variable Speed Constant Frequency, VSCF (such as the dc link and cyclo-converter systems).

3. 270 Vdc Systems, as proposed by NADC.

4. Variable Frequency Constant Voltage, VFCV, as used on some helicopter and early fighter aircraft.

5. Special (nonstandard) Systems, such as the

    - Variable Voltage/Variable Frequency (as proposed in earlier NASA/Lockheed studies)

    - High Frequency (20 kHz) Transmission Systems (as proposed by NASA-Lewis Research Center)

6. Dedicated Auxiliary Power Units.

The first three candidates are considered viable to the degree that the R&D is significant and implementation has, or is, being effected. Candidate No. 4 has the merits of simplicity and reliability, but the application has been limited to small (11 to 30 kVA) capacity systems, (e.g., the F-104) or to the large capacity systems on turboprop aircraft, such as on the P-3 aircraft. In

the Electra/P-3 aircraft, the engines (Allison T56) operate at constant speed so, to this extent, this direct driven generator system is another type of CSCF system. Candidate No. 3 is expected by the U.S. Navy to bring about an improvement in the power generation systems by the simple artifice of rectifying the full output of each 200/115 V three-phase generator with a three-phase bridge rectifier: problems in this system, however, reside in the switchgear problems of 270 Vdc and the lack of development (availability) of 270 Vdc equipment. Candidate 5 each have characteristic merits and are in need of further evaluation: the VV/VF utilizes presently developed components and can utilize different types of static power conversion. The 20 kHz system is a new entry into the APGS technology field and is presently in need of detailed evaluation, with respect to hardware, power transmission lines, etc. Candidate 6 is often proposed as a means of divorcing the secondary power system demands entirely from the propulsion system, but there are installation and other disadvantages.

3.2.1 Constant speed drives. - It is clear that the CSDs, in their various development phases, have met the needs of the electric power systems in most military and commercial aircraft over the last thirty or more years. These CSDs were developed to drive generators, designed to specification such as MIL-G-6099/MIL-G-21480, and to furnish power to utilization equipment in accordance with MIL-STD-704B and later MIL-STD-704 specifications. Almost exclusively, these drives were of hydraulic-mechanical design, using back-to-back piston type hydraulic pump/motor elements (Sundstrand), although GE developed a ball-pump CSD, which had limited in-field service application. Another concept, was the fully-mechanical type of drive that was developed by the Lycoming Company. An updated version of this drive by Lucas Aerospace Company is now used in the U.S. Marine Harrier aircraft, the AV-8A; it is an 11/20 kVA capacity system.

In the hydromechanical types of drive, the operational principles of the piston and ball pump designs are the same: the earlier (Sundstrand) versions consisted of a variable-displacement hydraulic pump, ported directly to a similarly sized hydraulic motor. The hydraulic motor was geared directly to the generator output and, consequently, had to be sized to the torque capacity of the generator. To this extent, these earlier drives were 100 percent hydraulic to mechanical power conversion systems compared to the later Sundstrand drives, in which the hydraulic elements were sized to make up the difference between the synchronous power output and the variable power (mechanical) input. These CSDs became known as the AGD (axial gear differential) and, later, the IDG (integrated drive generator). In these later drives, the hydraulic motor, operated into a planetary (differential) gear system, in which the hydraulic motor output shaft rotated at a speed, and in a direction, to maintain a constant output speed for the generator. Speed control of the motor was exercised, in the Sundstrand drive, by a swashplate and, in the case of the GE ball-pump drive, by an eccentric cam.

Some of the early development problems of the hydromechanical drives resided in the high Hertzian stress levels, lubricity problems, close mechanical tolerances, hardening/finishing processes, hydraulic-seal problems and thermal management. Overall, the problem of providing high torque/constant speed output over a variable input speed range is not simple, and the successful development of the aircraft type CSDs has been a pervasive problem.

For many years CSDs were a practical solution to the engine speed range problem and in the ensuing development their size and specific weight have progressively decreased, making it difficult for the all electric VSCF systems to complete purely on the basis of weight.

The IDG side-by-side configuration (figure 22) has been the most recent development in CSD technology and it has been selected for the Boeing 757/767 aircraft. It is a revised configuration of IDG that retains most of the features of the in-line IDG, such as a spray oil-cooled (SOC) generator and similar hydraulic log elements, with improvements in packaging and hydraulic cooling/lubrication. Overall, the new drive offers an improved, more compact, mechanical configuration which is projected to offer reduced life cycle costs and lower maintenance support costs. For the future, Sundstrand will evaluate higher power level drives and higher speed drives.

Thermal management and overall efficiency of CSDs (and VSCF systems) are important, so heat exchangers must be sized to handle the maximum heat dissipation, under the most adverse secondary cooling loop conditions.

CSD generators are designed to provide MIL-STD-704 electrical power quality and performance. One performance requirement relates to voltage and frequency recovery, when high loads are dumped on and off the generator. The CSD typically uses an isochronous speed control (with no speed-droop with load), but it relies upon a closed loop hydraulic servo and electro magnetic governor head trimming. When a sudden load is applied to the generator there is a momentary droop in speed until the servo-loop corrects for the deviation by an adjustment to the hydraulic/spool-valve. Thus the performance characteristic of the drive depends upon the loop gain, loop dynamics, and the inertia constants of the drive.

The paralleling requirement for CSD increases sophistication of the control functions, since real load and reactive load sharing must be effected through current transformers, operating into static control circuits. Typically, the minor adjustments are necessary to correct the minor offspeed

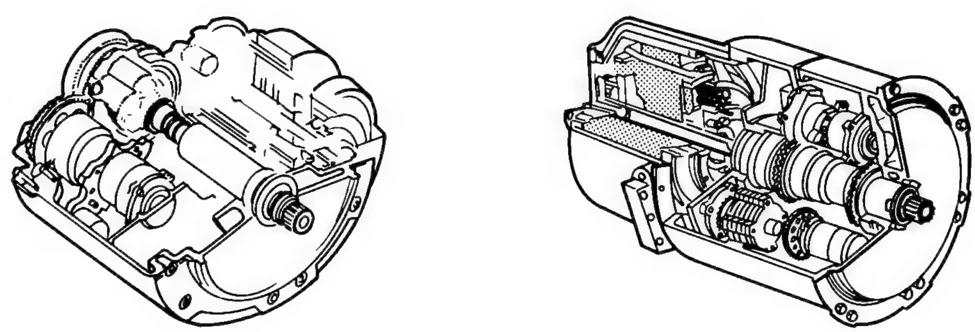


Figure 22. - CSD configuraton.

conditions (which affect real load sharing), and generator field excitation (which affects the reactive load sharing). These adjustments have in many drives been made through a magnetic trimming-head, which provides a vernier adjustment to the hydraulic spool valve. Some instability problems are possible and can occur in paralleled CSD installations due to differences in the closed loop response. This type of instability is usually more manifest during generator synchronization and paralleling. Such perturbations, when they occur, result in "load-swapping", and the instability remains until the system achieves stable real load sharing conditions. In the event of an overspeeding (overfrequency) in one CSD channel, the other power channels receive "reverse-real-power" and the overspeeding generator "hogs" the load on the paralleled bus. During this anomalous operating condition, the "normal-speed" generators run as synchronous motors above the speeds of their respective drives (via their overrunning-clutches). The modus operandi of the CSD protection system in this failure mode is to return the system to nonparalleled operation and to put the overspeeding drive underspeed condition; at this point the bus contactor, in the deviant power channel, is opened by the under-frequency protection.

A parameter that distinguishes the CSD from the VSCF type systems is the method of "rating". Normally, a drive has a rating that is based upon a "cubic-mean loading", which means that the service life (TBO warranties etc.), are dependent upon a definition of the load/speed schedule that will prevail in typical service. Thus the wear on the drive and its life/TBO statistics, are dependent upon this mechanical loading cycle.

Assessment of the reliability of CSD systems is best made with respect to specific applications, since there is a wide scatter of reliability data, relative to different airplanes. Basically, CSD/IDG/IDGS systems are scheduled for "on-condition" maintenance, except for servicing type inspections: oil must be checked say every 200 hours, which should take a few minutes, after access is provided. Pressurized oil-refills are necessary every say 700 hours and this procedure may take approximately 6 to 8 minutes. Other typical inspections include checking the drives for leakage and for clogged oil filters, which should take 2 to 3 minutes.

3.2.2 VSCF technology. - The acronym VSCF is the generic reference given to power generation systems in which the constant frequency power is derived from static power converters that are furnished with variable speed/variable frequency power direct driven generators. It therefore relates to any system in which the variable frequency output of the generator is converted to constant frequency or constant voltage dc power. Typically, the output of the generators can be rectified within the generator housing or, it may be rectified remotely in a static power converter package. Within this context, the following are evaluated as VSCF candidates:

- Cycloconverter VSCF: ac to ac power conversion
- DC Link VSCF: dc to ac power conversion
- 270 Vdc (VSCF): ac to dc power conversion

Figure 23 shows an artist's schematic illustration of the VSCF cycloconverter and dc link technologies.

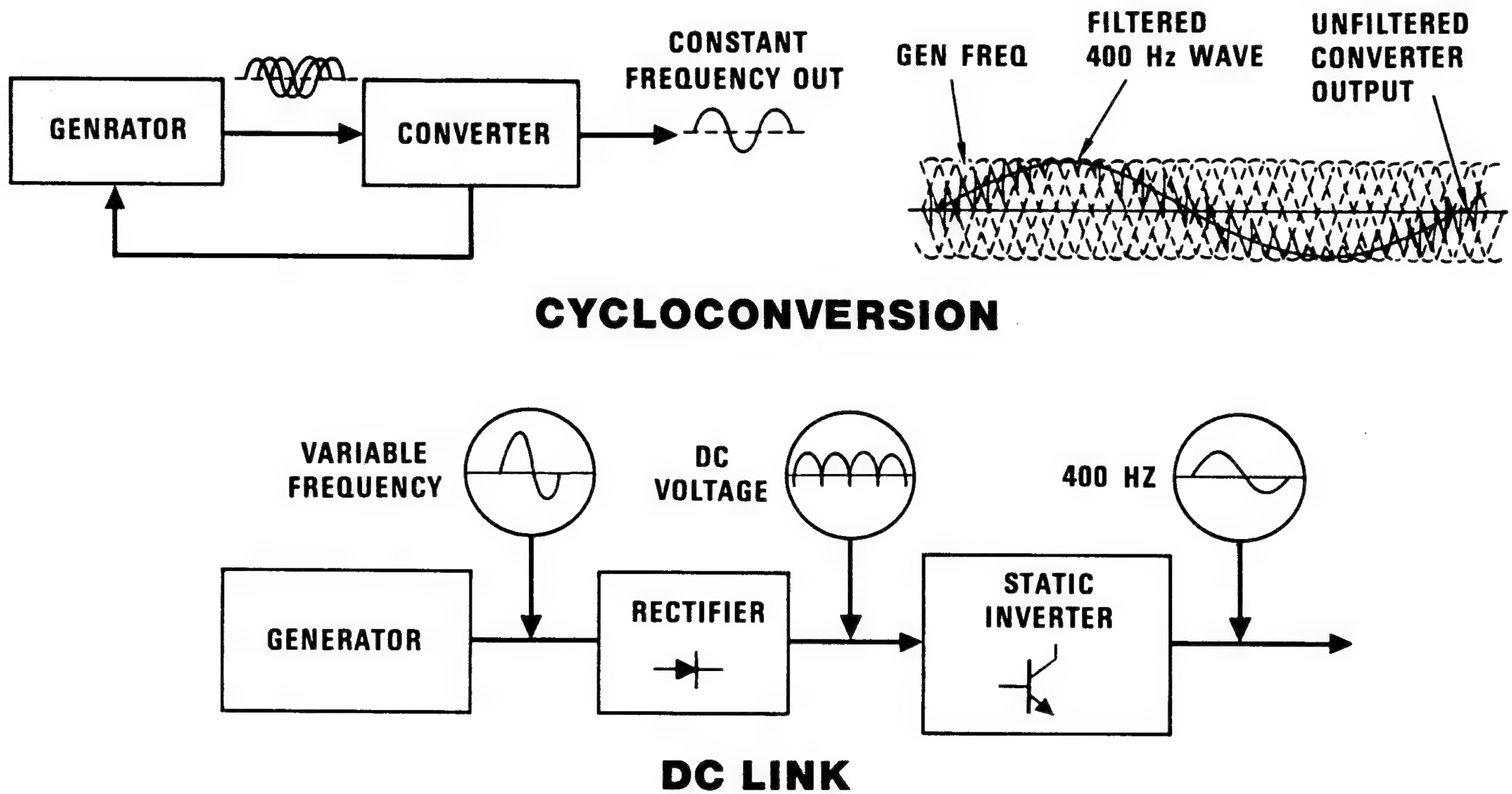


Figure 23. - VSCF cycloconversion and dc link.

The cycloconverter VSCF has the advantage of being the beneficiary of the most R&D development and it has been adopted for the A-4D, A-10 and F-18 aircraft. The U.S. Air Force and U.S. Navy have also funded programs directed to the development of VSCF systems in capacities of 40, 90 and 150 kVA. These VSCF generators may be wound rotor or permanent magnet type, but recently samarium-cobalt type rotors have been used with the converters that have been designed with a reverse-frequency programming-control, that permits the generator to function (in the reverse mode) as a synchronous starter motor. The first production type aircraft installation of this type, in which electric generator is to be used as an engine starter, is the 40 kVA installation for the U.S. Air Force A-10 aircraft.

Filtering is critical to the VSCF power systems and improved capacitors have been highlighted as key development items. Highly reliable filter capacitors with low specific volume are also a continuing activity in the development cycle of the VSCF power systems. The parts count is also key to the reliability and physical characteristics of the electronic assemblies, as are the electrical ratings of each of the components. While electric/electronic devices exhibit no physical wear characteristics, they are electro-thermally sensitive and voltage surges/spikes pose a threat to the junctions of the solid state power devices. Inadequately rated devices can therefore inhibit reliability along with any inadequate implementation of the cooling provisions.

A characteristic requirement of the cycloconverter system is that the input frequency to the converter (even at the minimum speed of the generator) be not less than three to four times the output frequency. This means that for a 400 Hz output, the frequency at base speed must not be less than 1200 to 1600 Hz. A legacy of this is that the generator must be driven at high speed and be designed with a large number of poles. Typical pad speed requirements for VSCF generators are therefore 9500 to 21,000 rpm. Another characteristic endemic to the cycloconverter is the ratio of the generator-kVA to the bus-kVA. Typically the generator kVA is about 1.40 to 1.5 times the bus kVA. Also, like all constant power VSCF type power systems, the generator must furnish full kVA output down to one half maximum speed of the engine. Because of this, the generator, electromagnetically, is capable of twice the output at the takeoff cruise speeds of the engines and consequently it is heavier than the CSD driven generators.

Cycloconverter VSCF systems are available in "integral" or "split" systems. In the former case, the electronics are accommodated within the generator housing to take advantage of the oil cooling system provided for the generator. Figure 24 shows the rotor removed from its stator. It is normally press-fitted (for good thermal-conduction) into the tunnel of the housing. Oil through the housing allows an outward heat-transfer from the generator stators windings and an inward heat-transfer from the electronics. In "split" type systems the electronics are separated from the generator and are accommodated in a package which may be located in the wing root area, or somewhere in the fuselage. Optional cooling is available for the electronic converter package and, although liquid loop cooling can be used, flow-through air cooling is more typical. Air cooling itself may be provided by discharge air from the cabin or by outside fan-propelled air on the ground. Air cooling

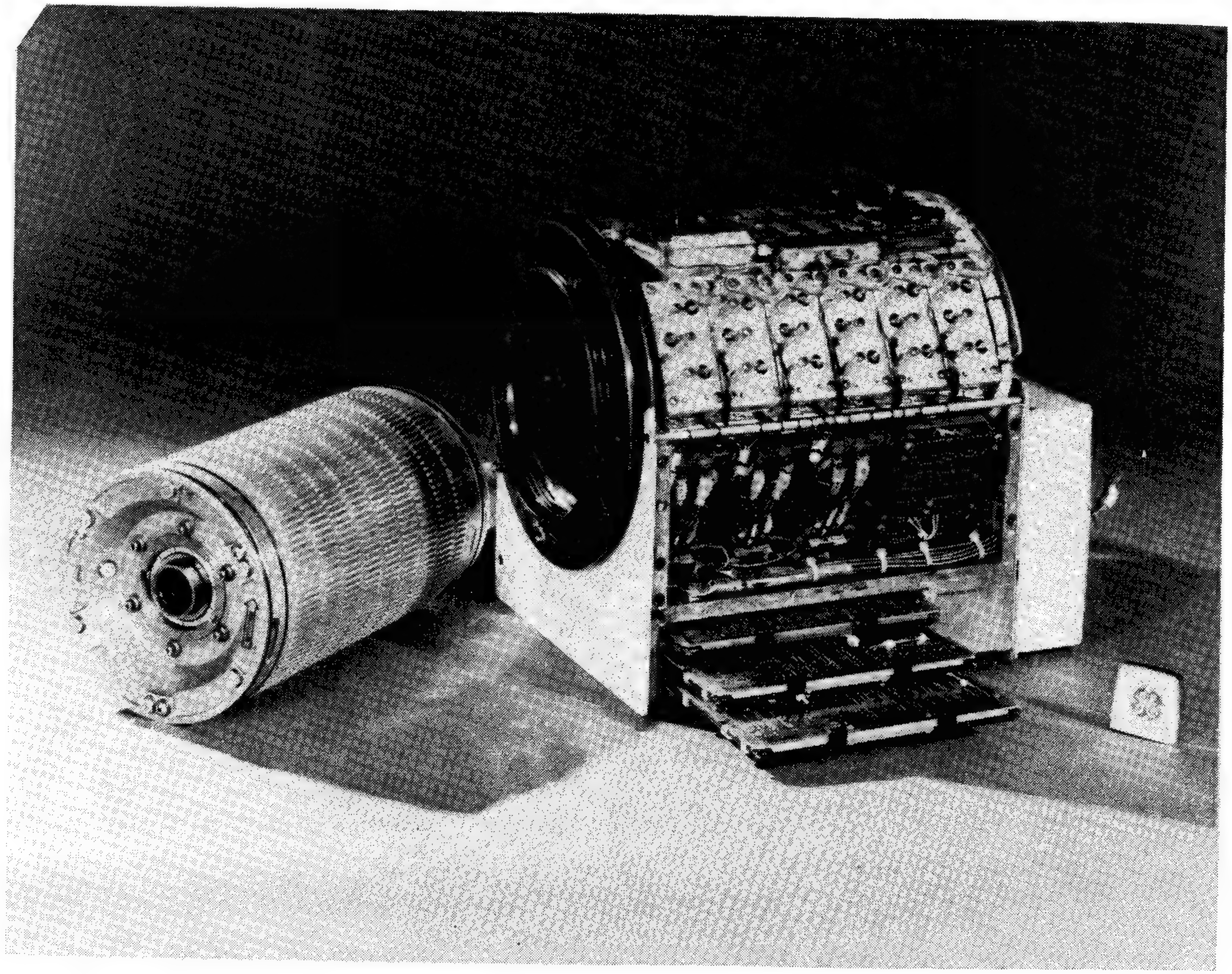


Figure 24. - F-18 VSCF integral generator/electronics package.

tends not to be as effective as liquid cold plates, but the air cooling can provide higher reliability. Also there are more sophisticated cooling methods that could be considered in advanced aircraft designs. Cooling technology could include evaporatively-cooled cold walls, as proposed by Lockheed, by immersing the electronic components in a thermally conductive fluid, such as fluorocarbon (Fluorinert) liquid bath. This fluid is cooled by a novel Freon evaporative cooling device and expansion bladder configuruation. (See figure 25.)

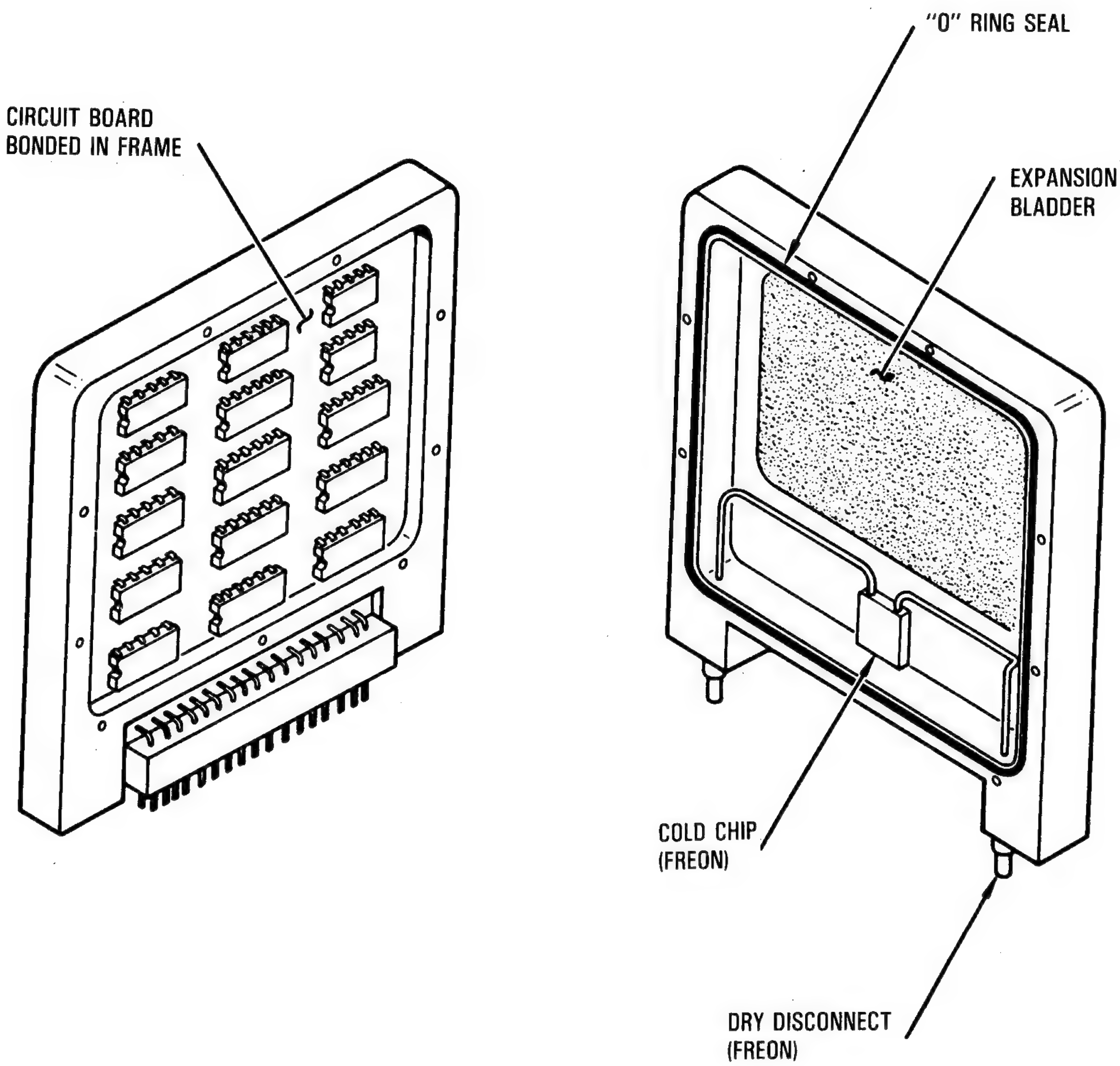


Figure 25. - Single board cold bath module.

An important engine consideration with the VSCF type power system is the requirement for high speed pads that must be provided on the accessory gearbox. To provide a less complex, lighter system, the oil cooling of the VSCF system in the power plant should be integrated with the engine oil cooling system. The engine manufacturer assumes responsibility for any integrated oil cooling system and priority would be given to the engine oil system in the event of any oil leakage problems in the generator. If the engine supplier cannot be persuaded to accept the VSCF generator cooling responsibility, then the provisioning of an autonomous pumping system would impose undue hydromechanical complexity on the VSCF power plant installation. Up to this time, the engine manufacturers have resisted the concept of integrated cooling, but with the use of SmCo generators, there is no rotor heating and, therefore, no need for through-the-rotor cooling that requires dynamic seals. With this new technology change the engine manufacturer may now become more receptive to the idea of engine-integrated oil cooling systems.

The attributes of the VSCF systems are that they can provide high quality power for loads such as ASW electronics, ECM, and the typical avionic equipments involving air data/flight management and other sensitive avionic loads. The latter systems are becoming increasingly important in advanced commercial transports. There are presently no issued MIL-SPEC documents for VSCF (and the 270 Vdc systems), although under the technical direction and coordination by Naval Air System Command a new specification, MIL-E-23001/3D(AS), is being developed to cover the VSCF cycloconverter system.

In VSCF power systems, frequency regulation can be controlled to fractions of a percent and there are, theoretically, no transient frequency deviations with respect to load and speed changes. Phase voltages are individually regulated which significantly reduces phase voltage unbalance with any unequal loading of the generator phases. It is however in the areas of reliability/maintainability and life-cycle costs that the VSCF suppliers claim to have their main advantage. It is argued that aircraft generators are already highly developed and, because the static power-converters have no moving parts, the reliability and maintenance support of the VSCF systems is projected to be better then hydromechanical electric systems. However, this may be simplistic to the extent that as stated, the highest reliability can be achieved only if the generator system supplier exercises diligent care in the thermal management of the electronics and conservative electrical ratings are used for the semiconductors, etc. Based on such criteria being met, it is possible that the VSCF system life cycle costs could be lower than CSDs.

As a continuing development activity the VSCF suppliers must work towards a reduction in the parts count and to developing improved transistors/SCRs, diodes, and capacitors. Some earlier VSCF converters had component counts in excess of 2000 and this worked against the objective of high reliability. With fewer parts and improved solid state technology, better performance, weights and reliability can be projected for the VSCF power systems.

3.2.3 VSCF dc link. - Much of the technical overview given on the cycloconverter VSCF applies to the evaluation of dc link VSCF system. Basically, the dc link VSCF is a dc-to-ac constant-frequency/constant power system that "inverts" the dc power that is obtained by rectification of the generator's

variable voltage/variable frequency output. Again an integral design may be used, wherein power is rectified and contained within the generator envelope, or it may be a split system. In the latter case, the ac is rectified in the front-end and inversion is accomplished in a remotely located power electronics assembly. With the remote static assembly, cooling may be separately provided by a dynamic cooling loop of the liquid type or forced air cooling may be adopted (depending on the installation preferences of the aircraft company or the design dictates of the equipment supplier).

Power quality of the dc link VSCF system is comparable with the cycloconverter system, but the inverter requires input and output filters unlike the cycloconverter, which uses an output filter only. However, the dc inverter is less complex in that it uses fewer solid state components, e.g., only six full wave power transistors versus thirty-six SCR (six phase FW bridge). Also the power losses in the bipolar devices are less than the SCR. Finally, the dc link does not suffer the generator-to-bus kVA rating penalty of the cycloconverter, but it does require a neutral forming transformer on the output. One other advantage of the dc link inverter is that the recirculation of reactive kilovars is constrained to the output of the inverter. Again, phases are individually regulated in this type of VCSF system and frequency regulation is dictated by the frequency reference standard. The unique difference of the dc link VCSF is that there is no four-to-one relationship betweeen the converter output frequency and the generator frequency. Therefore, high frequency (viz: 1.6 to 2 kHz) power transmission (that necessitates shielded cables) is not a prerequisite of this system. As a consequence, conventional frequencies such as 400 Hz, or the less conventional frequencies of 600, 800 Hz and higher could be selected as the nominal power transmission frequency (at the normal engine speed conditions). These frequencies could be achieved by pole/speed combinations as follows:

400 Hz: 4 poles/12,000 rpm or 2 poles/24,000 rpm

600 Hz: 6 poles/12,000 rpm or 4 poles/18,000 rpm, 2 poles/36,000 rpm

800 Hz: 6 poles/16,000 rpm, 4 poles/24,000 rpm, 2 poles/48,000 rpm

1200 Hz: 6 poles/24,000 rpm, 4 poles/36,000 rpm, 2 poles/72,000 rpm.

In summary, the claimed advantages of the dc link VSCF are the localization of the recirculating kilovars (with nonunity power-factor loads); the lower stress on the power components; the few parts count and the fact that the generator kVA rating is only about 1.16 times to bus kVA rating. Presently there are about three aircraft applications of the dc link system (the F-16, AV-8B, F-5G or F-20). It is of note that until recently the capacity of the dc link VSCF systems was limited by the transistors to about 40 kVA.

3.2.4 270 Vdc (VSCF). - While not normally considered a VSCF system, the 270 Vdc power system does fall into the category of VSCF systems, since it utilizes variable-frequency power and it does obtain the power from direct driven generators of the wound-rotor or SmCo type. The Navy's (NADC) sponsorship of a 270 Vdc development was prompted by their concern for the high maintenance support costs of other systems and the complexity of generator

paralleling. NADC's selection of 270 Vdc, vis-a-vis a cycloconverter on dc link VSCF system, was also based on the premise that the 270 Vdc power could be easily obtained without the use of complex electronic converters simply by rectification of any three-phase 200 Vac power supply. In this latter regard the ability to obtain 270 Vdc on carrier aircraft by the rectification of 3-phase 200 V 60 Hz power was an influencing consideration.

With three-phase 200/115V 400 Hz power, 270 Vdc can be obtained by the use of a full wave bridge rectifier, the relationship of the dc output voltage to the ac input voltage being related as follows:

$$E_{DC} = \sqrt{2}\ E_L \frac{m}{\pi} \sin \frac{\pi}{m} \tag{1}$$

where:

E_L = line-to-line ac voltage

m = number of phases ($\frac{\pi}{m}$ is expressed in radians)

Thus, a rectification of three phase power (by a full-wave bridge-rectifier, having six effective phases) results in a fundamental ripple of 2400 Hz and the dc output voltage is derived as follows:

$$E_{DC} = \sqrt{2}\ 200 \frac{6}{\pi} \sin \frac{\pi}{6} = 270.095 \text{ volts}$$

The conversion to 270 Vdc is therefore simply accomplished with a constant voltage three-phase 200 V input power supply, but the basic simplicity of the total system is compromised by the fact that fairly large amounts of 200/115 V 400 Hz are required, and these power supplies must be provided by sophisticated electronic inverters. Also, since the generator operates over a typical 2:1 speed range, the voltage of the generator is not constant, so phase-angle control of the full wave SCR bridge must be employed to harness the ac generator voltage at takeoff and cruise speeds of the engines. Thus, in the event of a regulator failure (or on short-circuit on the generator output), the 270 Vdc output will surge towards 540 Vdc and higher, thereby impressing a high recovery voltage across any fault interruption device. As in all constant power VSCF systems, the direct driven generators engender a 100 percent weight penalty in terms of their electromagnet (relative to the generators driven by CSDs).

The main shortcomings of the 270 Vdc power system are that the dc power cannot be used to power directly the very many ac and dc motors found in a large military or commercial transport-type aircraft. In such aircraft there may typically be more than 100 to 150 motors some of which, as in the IDEA/Alternate IDEA aircraft, might be over 100 hp. Brushless dc motors require static power inverters to furnish them with synthesized ac power, so the size, rating, weight and cost of these inverters will be related to the

horsepower requirement. As an example, a 25 hp Freon compressor motor would require approximately a 25 kVA capacity inverter and a cabin compressor drive motor, say 125 kVA inverter.

The development of high rupturing capacity solid state or hybrid power-controllers are the other aspects of the 270 Vdc system concept. Natural arc-extinction is not present with dc switching (as it is in ac switchgear), so this presents a problem at high altitudes. Also, as stated, generators, designed for a 2:1 speed range will have prospectively high recovery voltages on release of faults when they are operating at full speed. Switchgear design and circuit rupturing capacity on short circuits are therefore key to the complete maturation of the 270 Vdc system. Further, the full and efficient application of 270 Vdc cannot take place until there is a conversion of the many present loads over to 270 Vdc. The development of 270 Vdc switchgear and 270 Vdc utilization equipment must therefore be aggressively pursued to ensure the orderly development of this type of power generation system.

An alternative which is considered attractive for the application of the 270 Vdc technology in large capacity power systems projected for the IDEA/ Alternate IDEA type airplanes is a hybrid three-phase ac/270 Vdc system. In this type of APGS, ac power is derived on a dedicated basis for loads where constant power over the speed range is required. Such critical loads would be the primary flight control surfaces in the IDEA.

From a performance standpoint assessment the 270 Vdc generator system is an essentially simple and reliable generator concept. The power quality is good, and the high frequency ripple components can be relatively easily filtered, to provide a constant output level of voltage. On the other hand, the quality of the power delivered to many of the equipments not powered directly by the 270 Vdc will be dictated and determined by the sophistication of the inverters that provides this power. Similarly, dc to dc converters providing 270 Vdc to 28 Vdc (and other dc voltages) will have power quality inherent in the particular converters. There is no problem of phase voltage unbalance, etc., but at typical flight operational speeds, the output voltage of the generator must be harnessed by the use of phase angle control of a full wave rectifier bridge. Inherently the 270 Vdc generator is capable of providing 540 Vdc at normal flight engine speeds, so under these conditions the SCR phase angles are delayed to limit the value to 270 Vdc. On the ground (at 50 percent speed) however the generator capacity and the SCR phase angles are advanced to a point where all of the area in each half cycle of the generated sinusoidal wave is utilized.

3.2.5 VSCF technology: summary. - The foregoing evaluation of VSCF technologies shows that there are no major clear cut advantages between the cycloconverter VSCF, the dc link VSCF, and the 270 Vdc systems. The cycloconverter VSCF has an impressive development genealogy and it has been developed in capacities up to 150 kVA and higher. Its performance has also been impressively demonstrated by the starting of engines up to the 40,000 pound thrust category. In addition, VSCF cycloconversion has also been the beneficiary of extensive laboratory experience and a large amount of flight experience via USAF/USN development programs. A salient difference between the VSCF dc link is that it does not require especially high frequencies and, therefore, does

not require very high generator speeds and the use of shielded-cables to mitigate the EMI problems. It has however been limited, presently, to the lower capacity systems, as in the F-5G and AV-8B applications. The dc link like any VSCF system still needs high generator speeds (to be weight competitive with CSD systems) and, like the cycloconverter system, it requires an autonomous oil cooling system (one that is independent of the engine oil cooling system). The problem of furnishing oil remains one of the primary concerns for the VSCF/270 Vdc power systems: the alternative is to use blast air cooled generators. If step-up gearboxes are necessary it is possible that wet sumps with splash-lubricated gears could be used and the gearboxes could take advantage of conducted/radiated cooling. A 90 kW gearbox with 2 percent efficiency loss would dissipate some 1.84 kW (approximately 104 Btu/min.). This would not pose a thermal management problem. However, the need for a step-up gearbox will be a penalty only if a high speed pad is not provided on the engine and geared oil pumps are not required. In the IDEA/Alternate IDEA it is projected that the engines will be designed to furnish the necessary high generator speeds and integrated oil cooling.

One of the prospective advantages of dc link VSCF or a hybrid ac/270 Vdc system is the ability to use partial power conversion since, in most aircraft applications, there are no large demands for precisely controlled power on the ground. Under most operational conditions the turbofan/turbojet engines operate in flight at a relatively constant speed (of say 92 percent), so the nominal (400, 600, 800 Hz) power supply could be used directly for the many utility loads, such as the galleys, heating, lighting, de-icing, and many ac motor loads.

A typical weight disadvantage of the VSCF system vis-a-vis the CSD systems is that the generator provides 100 percent power over the 2:1 speed range. The CSD does this also, but the VSCF system is better able to adapt to a power-proportional-to-speed concept to give it a weight advantage. As it is, a VSCF generator designed for 150 kVA at 50 percent engine speed becomes electromagnetically equivalent to a 300 kVA machine at maximum speed. Also, like the 270 Vdc system, it has the potential for generating high voltages at maximum speed when it is designed to meet the overload requirements for MIL-G-21480 and prevailing generator specifications such as MIL-E-23001/3D-(AS).

The 270 Vdc system presently lacks in-field service experience and suffers from the unavailability of a substantial inventory of 270 Vdc equipment. Also it cannot power motors directly. The other primary shortcoming is the problem of switchgear and circuit breakers/RPCs/SSPCs in handling prospectively high fault currents at high altitudes. The 270 Vdc system could, however, become a viable alternative when the switching technology improves and more 270 Vdc equipment becomes available in the marketplace. In the interim a conventional (VSCF) 200V/400 Hz and say a 400V/800 Hz type power system could still provide 270 Vdc for any special 270 Vdc equipments simply by local rectification of the ac in those circuits. Thus, a nominal 270 Vdc power system could become a hybrid system with the capability of providing both 200 Vac and 270 Vdc power.

Tables 6 and 7 are one supplier's comparison and breakdown of the physical weight, size, characteristics, etc., of a VSCFs cycloconverter and dc link

TABLE 6. ELECTRIC POWER SYSTEM WITH VSCF CYCLOCONVERTER MAIN CHANNELS

Item	Est. Wt.(lb)	Est. Size (inches)	Est. MTBF (hours)	Qty./ A/C	Total Est. Wt. (lb)	Remarks
Cycloconverter Generator	65.0	12H x9W x 17L	9,200	3	195	--
Cycloconverter	80.0	1,800 in^3	4,000	3	240	—
Main Channel GCU	10.0	12.6 x 7.6 x 4.9	21,600	3	30	Assumed same for dc Link System
APU Generator	45.0	9D x 10L	16,900	1	45	Assumed same for dc Link System
BP/APU Cont. Unit	12.0	12.5 x 7.6 x 4.9	15,200	1	12	Assumed same for dc Link System
Control CT Package	2.5	7 x 5 x 3.9	500,000	4	10	Assumed same for dc Link System
DPCT Package	1.5	6.6 x 2.8 x 3.9	1,000,000	10	15	Assumed same for dc Link System
QAD	4.0	--	--	4	16	Assumed same for dc Link System
			Total Component Weight		563	

Estimated System efficiency Over Speed range @ 90 kVA (excluding Gen.-Conv. Feeders): 73-77%

Estimated Gen-Conv. Feeder Weight: 1.85 lb/ft
Estimated Gen-Conv. Feeder Size: 6 Phase, -2 #6/Phase, 2 - #6 for neutral, and shield

Estimated Gen.-Conv. Feeder Losses @ 90 kVA: 43 watt/ft.

Estimated Coolant Flows, Main Engine Generator and Converter:
Generator: 2-4 gpm, 150°C Oil
Converter: 20 lbs/min. 30°C Air

TABLE 7. ELECTRIC POWER SYSTEM WITH VSCF DC-LINK MAIN CHANNELS

Item	Est. Wt.(lb)	Est. Size (inches)	Est. MTBF (hours)	Qty./ A/C	Total Est. Wt. (lb)	Remarks
DC Link Generator	57.0	12H x9W x 17L	9,000	3	171	--
Converter	85.0	1,800 in^3	18,000	3	255	Three parallel transistors
Main Channel	10.0	12.6 x 7.6 x 4.9	21,600	3	30	Assumed GCU same for dc Cycloconverter System
APU Generator	45.0	9D x 10L	16,900	1	45	Assumed same for dc Cycloconverter System
BP/APU Cont.	12.0	12.5 x 7.6 x 4.9	15,200	1	12	Assumed Unit same for dc Cycloconverter System
Control CT Package	2.5	7 x 5 x 3.9	500,000	4	10	Assumed same for dc Cycloconverter System
DPCT Package	1.5	6.6 x 2.8 x 3.9	1,000,000	10	15	Assumed same for dc Cycloconverter System
QAD	4.0	--	--	4	16	Assumed same for dc Cycloconverter System
			Total Component Weight		554	

Estimated System efficiency Over Speed range @ 90 kVA (excluding Gen.-Conv. Feeders): 78-82%

Estimated Gen-Conv. Feeder Weight: 0.9 lb/ft
Estimated Gen-Conv. Feeder Size: DC, 2 #6(+), 2 - #6(-) and shield

Estimated Gen.-Conv. Feeder Losses @ 90 kVA: 7 watt/ft.

Estimated Coolant Flows, Main Engine Generator and Converter:
Generator: 2-4 gpm, 150°C Oil
Converter: 20 lb/min 30°C Air

power system, based on a 90 kVA power channel. The tables show typical MTBF data and the lower portion of each table gives supplemental information, such as power feeder weights, power feeder losses and coolant requirements. Note: These data reflect a typical power generation channel and have no specific reference to the IDEA/Alternate IDEA. The information is, therefore, suitable only for a typical comparison of the cycloconverter and dc link VSCF system. Table 8 further amplifies the comparative characteristics of the two system except that the data shown in this table relate to a 30/40 kVA power channel.

The thrust of the foregoing tables is to highlight the advantages of the dc link over the cycloconverter, primarily in the area of the MTBFs (reliability). It also favors the dc link in the area of fewer power devices, higher efficiency, higher temperature tolerance, etc. In this regard, the supplier's bias enters into the evaluation so the verifiable facts must, in the final analysis, derive from the real world operational experience. This applies to all the physical, performance, thermal, reliability, cost factors and other data. For this reason, it is difficult to quantify all the factors, pursuant to the evaluation and analysis of all the candidate electric power systems for the IDEA and Alternate IDEA. Therefore, at the end of this section, the trade analysis is made qualitatively rather than quantitatively.

3.2.6 Variable voltage/variable frequency. - This system is different from the previous VSCF power generation systems in that it is a hybrid system, in which power derived from the engine driven generators is used directly to power many loads, while a small percentage of each generator output is converted to high quality power for avionics and other sophisticated equipment, such as FCS computers, etc.

By definition, a VV/VF system is a system in which the engine driven generator is either permanent magnet (SmCo type) machine, or a wound rotor machine (in which the field current is controlled, so that the voltage generated is proportional to frequency/speed). In such systems the V/F ratio is maintained at a constant value over the speed range of the engines so there is no concern for magnetic saturation. Also the stators of ac motors operate with essentially constant flux. This means that in the case of motors, the torque is also essentially constant over the speed range. Similarly transformers operate at constant flux over the voltage and frequency range so the primary/secondary voltage values are related typically to the turns-ratio. The equations that relate the electrical design of motors and transformers (electromagnetic equipment) are shown below

$$E = 4.44\ k_1\ k_2\ NF\ \Phi \times 10^{-8} \tag{1}$$

TABLE 8. - COMPARISONS BETWEEN 30/40 KVA CYCLOCONVERTER AND DC LINK SYSTEMS

		Cycloconverter	dc Link
OIT Continuous		80°C	120°C
Device Temp. Limitation		150°C	200°C
System weight, lb		77	79
Efficiency, %			
Lo Speed		73.3	78.1
Mid Speed		71.4	76.3
High Speed		67.6	72.9
Switching Method		Integral control with bank selection	Microprocessor controlled, Precalculated, stored program switching pattern
Switching Devices		36 SCRs	12 Transistors 12 Diodes
Generator		Six Phase	Three Phase-Rectified
MTBF as a Function of Load and OIT (Power Devices only)			
80°C	40 kW	4770	83,119
	45 kW	3987	80,749
	60 kW	2552	70,542
93°C	40 kW	3470	67,659
	45 kW	2837	65,746
	60 kW	1546	57,143
121°C	40 kW	1493	40,933
	45 kW	1132	39,635
	60 kW	478	33,047
125°C	40 kW	1292	37,554
	45 kW	1132	39,635
	60 kW	388	29,903

where:

k_1 = dispersion coefficient

k_2 = chording factor

N = number of turns

F = frequency in Hz

Φ = flux

$$T = \Phi I_2 \cos \theta \tag{2}$$

$$T = \frac{E_1}{F} \cdot \frac{skE_1}{R_2 + sj X_2} \quad \arctan^{-1} \frac{SX_2}{R_2} \tag{3}$$

where:

Φ = stator flux

I_2 = rotor current

R_2 = rotor resistance

X_2 = rotor reactance

s = slip

k = transformer ratio (stator/rotor)

These formulae are electromagnetic relationships: (1) shows the basic relationship that flux Φ is proportional to the E/F ratio. (2) shows that if the stator flux ($\Phi = E_1/F$) is constant then the rotor current I_2 is proportional to skE and inversely proportional to the rotor impedance, where skE = voltage induced in rotor and R + sjX is the rotor impedance (at the slip frequency of interest). Where the slip is very low i.e., when rotor is near synchronous speed, the sjX function is small and the rotor power factor is high. As shown in figure 21, the load torque and the pull-out torque characteristic of induction motors (operating with a constant E/F ratio) remains relatively constant over the speed range.

It is appropriate to consider higher voltages and higher frequencies for aircraft such as the IDEA and Alternate IDEA. The frequency dictates the speed at which generators must be driven and the speeds at which motors will run when supplied with different frequencies. Peripheral speed constraints limit the upper range of frequency while the consideration of the high-altitude environment places an upper limit on voltage. Regarding the latter, it seems prudent to consider voltages not higher than say 400 V or 600 V, if problems of corona and voltage flash are to be avoided. Even then, careful installation practices will be essential to avoid problems at altitudes over 30,000 feet.

Clearly, if the E/F ratio of 200 V/400 Hz is retained, then this establishes the value of the voltages at the higher frequency, e.g., 400 V/800 Hz, 600 V/1200 Hz, 800 V/1600 Hz. There is no mandate to maintain this 200 V/400 Hz ratio, but it is logical to consider double-voltage and triple-voltage standard systems. Further, it is possible, in some cases, to use three-phase, 200 V/400 Hz motors/transformers on three-phase 400 V/800 Hz power if due consideration is given to mechanical and electrical constraints. For example,

a conventional three-phase, six-pole motor, running at 7500 rpm on 400 Hz power, would run at about 14,700 rpm on 800 Hz power. This much higher speed would impact on the bearing capacity/life and it might be in excess of the limiting peripheral speed of the motor. Windage and friction losses would also increase.

Further design considerations are the iron and copper losses incident on higher frequencies. Iron losses are represented by hysteresis and eddy current losses, while copper losses are related to current density.

$$W_h = \eta F \ B^{1.6} \times 10^{-7} \text{ watts/cm}^3$$
$$= KF \ B^{1.6} \quad (4)$$

$$W_e = KF^2 B^2 \quad (5)$$

where:

W_h = hysteresis loss in watts

W_e = eddy current loss in watts

F = frequency in Hz

B = maximum magnetic induction

η = hysteretic constant

These formulae show that the eddy current losses are proportional to F^2 and B^2, so they increase at a faster rate than the hysteresis losses with (which vary according to F and $B^{1.6}$). These losses, with minor differences, relate to stator iron in motors and laminations in transformers, filters, etc.

The frequency selection therefore tends to be a more sensitive parameter than voltage; also, as outlined in the Power Distribution Section, it affects the current density distribution in electric power feeders. This phenomenon was defined as "skin-effect" wherein the current tends to crowd into the outer surfaces of the conductor, resulting in an inefficient use of the conductor cross section. Skin effect occurs because the inner core of the cable has a higher impedance and so it operates at a much lower current density. Because it is a frequency related phenomenon, skin-effect is exacerbated by the higher frequencies and the current transfers more and more to the outer surface of the conductor as the frequency increases.

The effective ac resistance of the cable therefore increases along with the cable reactance ($X_L = 2\pi fL$). Another side effect is that the cable power factor worsens at high frequencies and it is more manifest with the larger size (power-feeder type) cables.

From the above it can be seen that the selection of frequency and voltage for the VV/VF system must be constrained by the practical dictates of the utilization equipment. Clearly, a frequency such as 5 kHz would be too high for many ac motors, because too many poles would be required to bring the motor speeds down to usable levels. The formula relating the variables is

$$F = N \times P/60 \tag{6}$$

where

F = in Hertz, Hz

H = revolutions/min (rpm)

P = pairs of poles

Many fuel pumps, hydraulic pumps and other loads operate at speeds of 4000 to 5700 rpm so high ratio gear reductions would be necessary to bring the motor speed down to these low values. For example, a six-pole motor operating on five kHz supply would run at 5000 x 60/3 = 100,000 rpm, and would require a gear reduction of approximately 20:1; also, the stress engineer would have to be critically concerned with the balance and whirling speeds of the rotor.

For the VV/VF system, the practical dictates of the utilization equipment therefore would tend to drive the frequency reduction towards the double or triple standard frequency levels only (unless one were to accept a considerable amount of power conditioning, as would be the case with a very high frequency system). Evidently, the VSCF cycloconverter system requires a high generation frequency because the nature of the power conversion process dictates a generator frequency that is some four times higher than the output frequency. Thus, for a frequency of say 800 Hz (at one half engine speed), the generator frequency (at full speed) would be 2 x 800 x 4 = 3.6 kHz.

The selection of frequency in the VV/VF system is influenced by the desire to use as high a frequency as possible to drive turbomachinery at high speeds without the need for gearboxes; and, on the other hand, not end up with a frequency too high, that high gear reductions are necessary for many motor loads.

Based on these somewhat antithetical considerations, the following voltage/frequency ratios appear reasonable.

IDEA	Three-Phase 400 V 800 Hz (0.5 E/F ratio)
Alternate IDEA	Three-phase 400 V 800 Hz or three-phase 600 V 1200 Hz

These E/F values seem practical when the physical limitations on motor speed are recognized along with the problems of corona and low ionization potentials at high altitudes.

Since the VV/VF system proposed is a partial power conversion system, it may have two or more high quality power supplies derived from the primary power supply. Since these dedicated power supplies will be typically low capacity and comprise only a small percentage of the total installed power capacity, the power quality of these supplies can be controlled to very close limits. All the other VV/VF power from the generators is available for the ECS, heating, lighting, galley, de-icing and miscellaneous actuation loads.

Typical power specifications such as MIL-STD-704B govern the power-quality requirements for the 400 Hz, 270 Vdc and 28 Vdc utilization equipment; MIL-E-23001/3DAS (NASC, custodian) governs VSCF cycloconversion (does not address VSCF dc link). NASC is developing MIL-E-85583 (AS), a generic electric system spec (with slash drawings for Navy aircraft). The USAF, ASD/ENESS (WPAFB) is developing "MIL-PRIME Electric Power Subsystem, Aircraft." Specifications MIL-G-6099 and MIL-E-21480 relate to air cooled generator designs while MIL-G-25704 governs constant frequency (CSD) ac power systems. The SAE committee A-2Y continues to work on MIL-E-85583(AS) and will also develop a 270 Vdc specification.

The performance of the dedicated power supplies in the VV/VF system will therefore conform to the pertinent and prevailing specifications. An objective of the VV/VF system, outlined in the previous NASA-LaRC/Lockheed Study (NAS1-16199, May 82), was to use a PM generator design where the V/F ratio was preserved over the load and speed without the need for a voltage-regulator. To achieve this a different generator design having the following design characteristics would be necessary:

- Relatively large air gap
- Low synchronous-reactance, X_d
- Low negative-sequence reactance, X_2
- Low zero-sequence reactance, X_o
- Low I^2R losses in stator
- Low core and pole-face losses.

A generator of this type would not be weight optimized, but many of the electrical parameters would be improved. The increased air gap would itself yield favorable electrical characteristics. For example, voltage unbalance is a function of the negative and zero sequence reactance (X_2 and X_o) and resistances (R_2 and R_o). Both X_2 and X_o decrease with increase of air gap length, so the phase voltages will be less sensitive to load unbalance. The X_D, the direct axis synchronous reactance, is itself a strong function of air gap, and it decreases with increase in air gap length; so as the air gap length increases, the transient-voltage excursions will also decrease. Operating at

lower current densities (0.9 to 1.25 kA/cm^2) (6 to 8 kA/in^2) in the stator would result in improved insulation life and increased efficiency. Finally, pole-face losses which are exacerbated by small air gaps less than <0.025 in. would also decrease with a large gap of 0.06 in.

In a wound-rotor generator the I^2R losses in the field (and exciter) increase. With a PM rotor there are no (rotor) field losses but the rotor magnet weight would be increased with the larger air gap.

The main legacy therefore of the proposed generator in the VV/VF system is that it would be heavier than a conventional high synchronous impedance machine, but the electrical performance will be inherently better. Whether the machine would operate without a voltage regulator from 0 to full load to overload is a matter for further study and evaluation. However, there are methods by which the voltage regulation could be constrained to low limits and one of these would be to establish the air gap flux for the per unit voltage at a quiescent load level, typical for the IDEA and Alternate IDEA airplanes. Fortunately, in these aircraft the all electric ECS, for example, reflects a constant (large) load to the generators so load changes will be relatively low, compared to a conventional airplane. Also, there is the possibility, as proposed by Lucas Aerospace, that a novel method could be used for controlling the voltage of the PM machine without chopping-up the output wave form with phase-controlled SCRs. These and other methods could therefore be explored in further evaluations of PM and would rotor generators operating with a constant E/F ratio characteristic.

3.2.7 <u>20 kHz power system</u>. - The system which is discussed in this section was conceived at the NASA-Lewis Research Center and the evaluation is based on a description of the system provided by the staff of NASA-Lewis. Figure 26 is a schematic presentation of a 20 kHz system applicable to a twin-engined aircraft showing two 40 kVA generators driven by each engine and four 3-25 kVA bidirectional converters. As shown, the four converters are paralleled via isolation devices (contactors or SSPCs) onto two high 20 kHz distribution buses that are routed around the aircraft. The possibility of using 60 Hz external power was described for engine starting, using what was, in effect, "half-converters." Each half converters is bidirectional in power flow, with an ability to "cross-start" (where one engine had failed in flight) by using one of the 1/2 converters on the other engine in a "power-sourcing" mode while the other 1/2 converter operated in a "power-receiving" mode.

The APU is eliminated on the basis that electrical (60 Hz) power could be used for engine starting rather than power from an onboard APU. Also, two "bidirectional-converters-with-charge-control" are used as the emergency back-up power supplies that could be individually power-sourced by batteries in the event of an all-engine-out failure. Under this mode, the batteries would furnish power to the converters which, in an inversion mode, would power either or both, redundant (HF) ac buses via appropriate dual power switches. A third switch in the output of each emergency inverter/converter is used for furnishing power to a noninterrupt ac power bus. Other information in figure 27 shows that power conditioning, for 20 kHz-to-400 Hz and 200 kHz-to-low voltage ac or dc are accomplished via two converters (synthesizers) fed from the dually redundant HF ac buses. Cabin pressurization for the airplane is

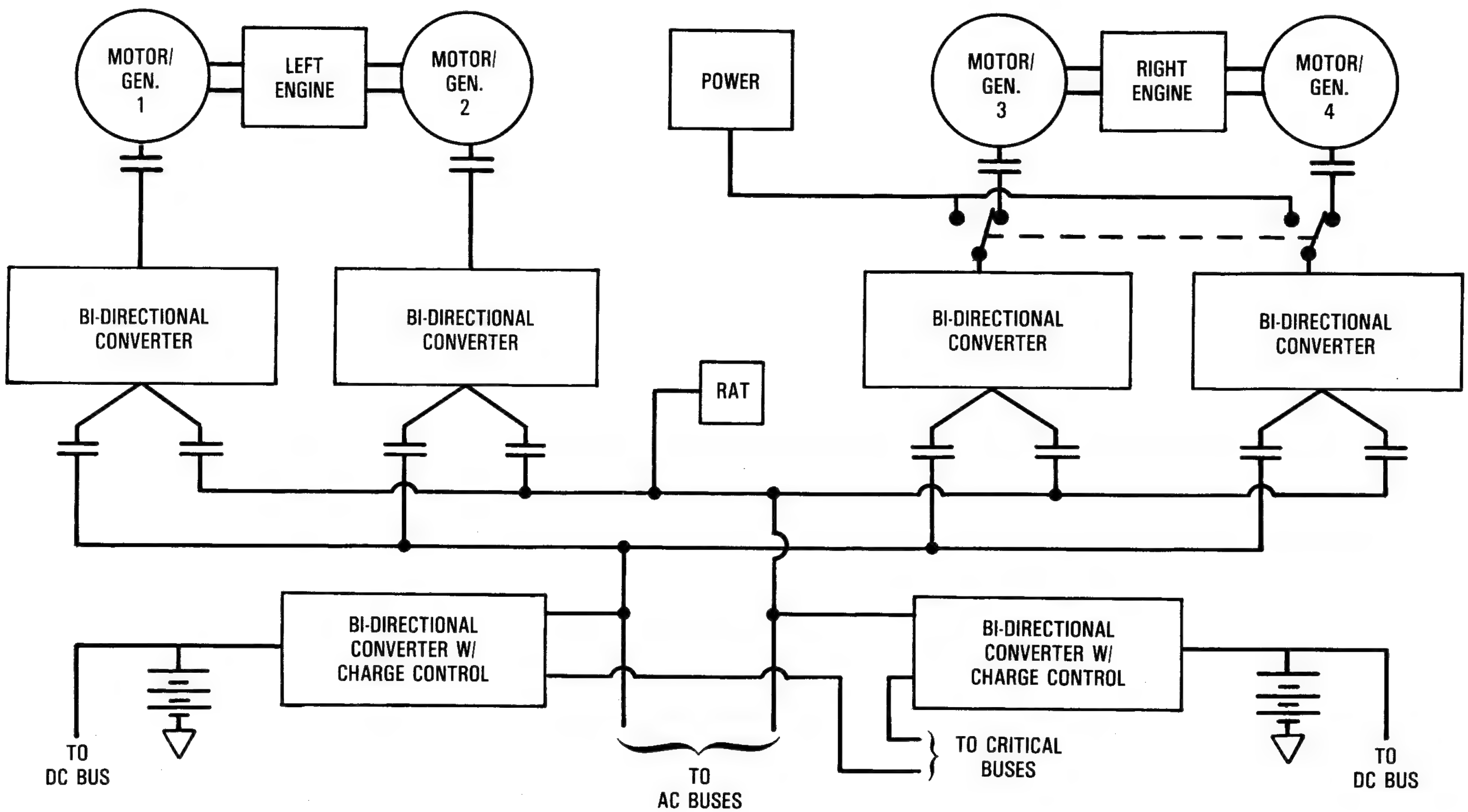
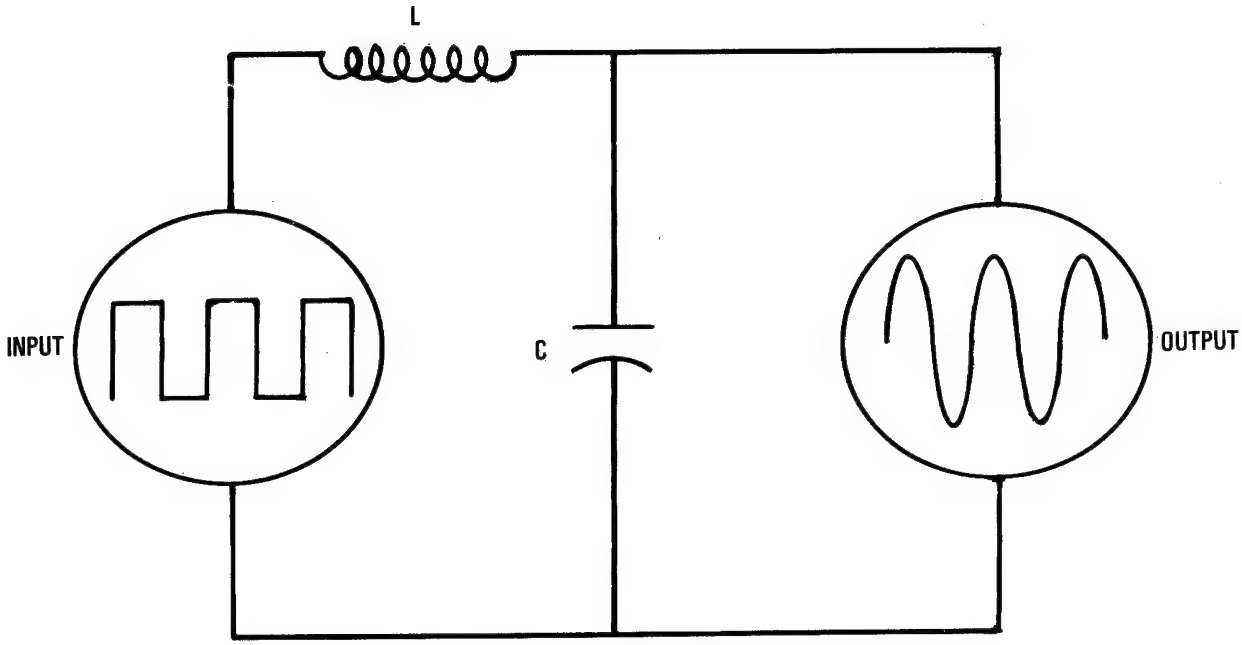


Figure 26. - 20 kHz power system.



RESONANT CONVERTER EQUIPMENT CIRCUIT
(VOLTAGE OUTPUT)

Figure 27. - Resonant converter equivalent circuit.

furnished by engine driven compressor, rather than electric driven compressors. The two ECS power-conditioned buses shown in figure 26 are used for Freon compressors, cooling fans, etc.

The basic theoretical principal of resonant power switching mode converters with simplified schematics of the type is shown in figures 27 through 30. These diagrams respectively show

Fig. 27 A square wave high frequency voltage having a fundamental resonant frequency (typically 20 kHz) that is supplied across a series LC network. The load in this case being in parallel with C, although the load could be in series with LC.

Fig. 28 The demodulation of a low frequency synthesized wave from the high frequency input wave.

Fig. 29 Transformer coupling of two resonant converters to illustrate bidirectional power flow mode of operation.

Fig. 30 Time-phased control of six switch bridge to synthesize three phase output at low frequency.

The above are basically illustrative of the 20 kHz power transmission systems and further discussion is given below.

There are many power conversion/inversion methods that involve dc to dc (converters), dc to ac (inverters), and ac to dc (converters). Well known types already discussed in this report are the VSCF power systems that are

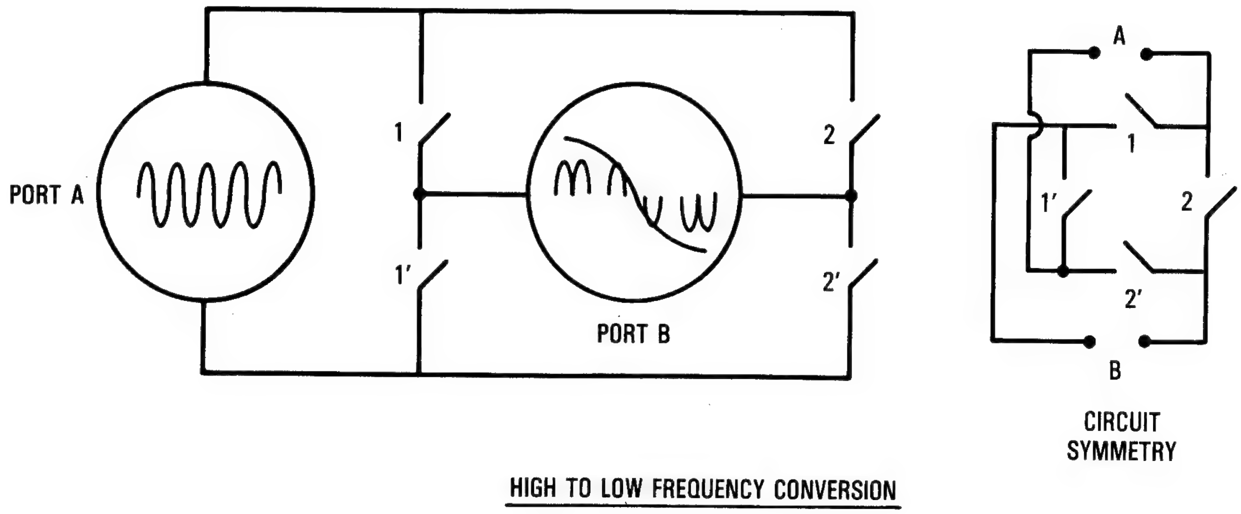


Figure 28. - High-to-low frequency conversion.

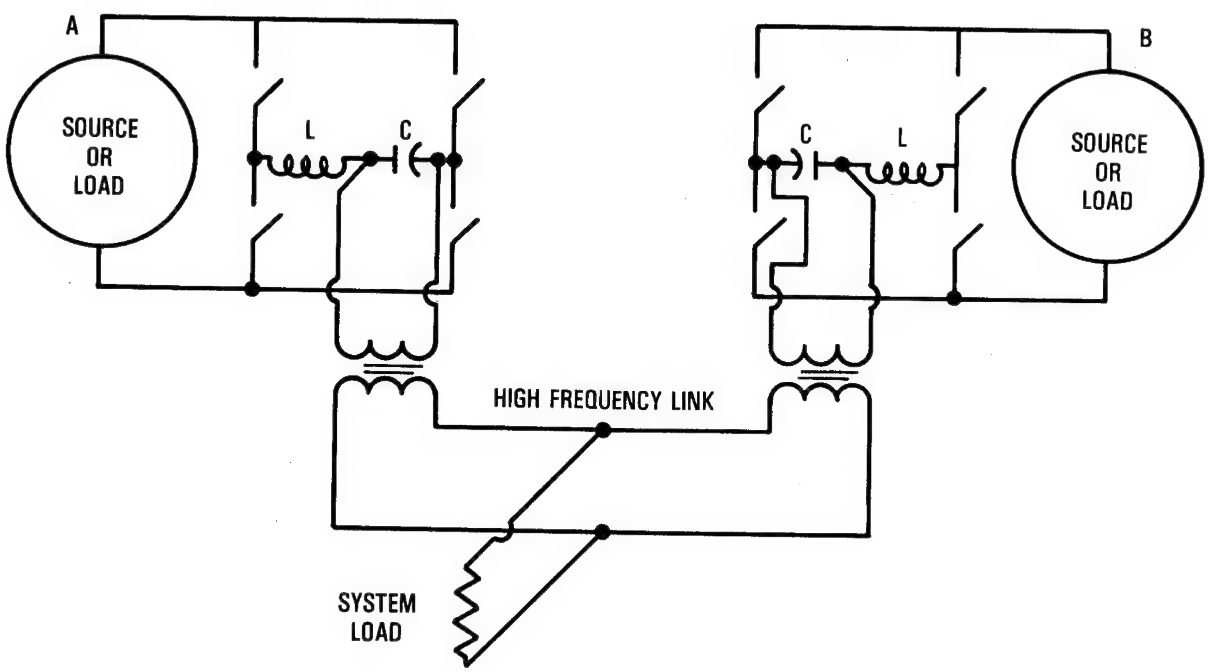


Figure 29. - Bidirectional high frequency power interface
(transformer coupled).

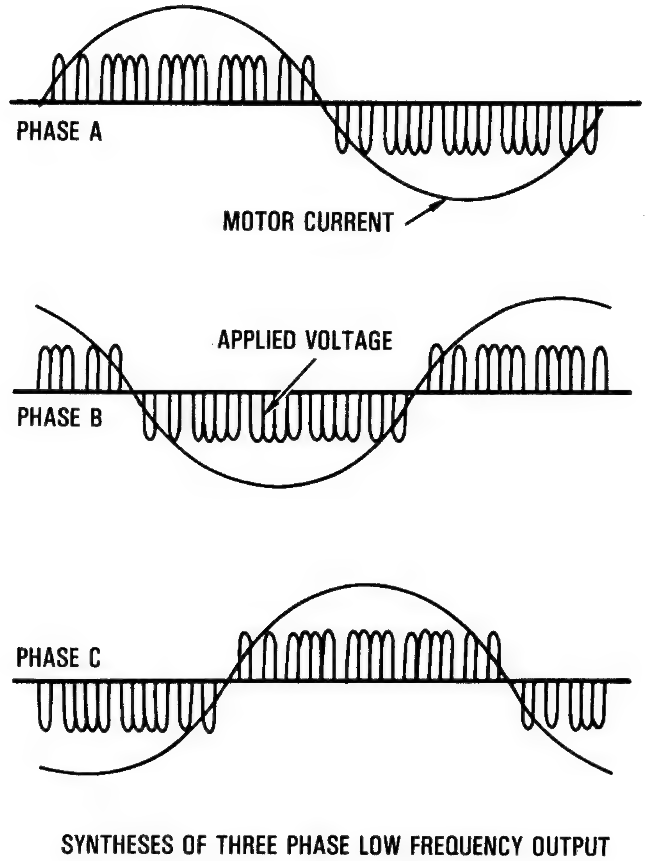


Figure 30. - Syntheses of three-phase low frequency output.

described as cycloconverter or dc link type systems. The 270 Vdc system is also a VSCF type system inasmuch as the generator is direct engine driven and develops variable voltage/ variable frequency power. In terms of inverter/converter technology the following summarizes some of the basic modes employed in high frequency switching converters/inverters.

- Cycloconverters (ac to ac); synthesizing a lower frequency waveform from a waveform some three to four times higher.

- DC link (dc to ac) inverters; accepting dc input (such as 270 Vdc), or rectified ac, which is commutated at some high switching frequency for conversion to ac.

- Switched mode power supplies, dc to dc, dc to ac, ac to dc, etc., where the objective is to achieve the transfer-function of an idealized transformer, i.e., the voltage/current waveforms on either side are faithfully reproduced in a lossless manner.

Figure 31 is illustrative of a switched mode ac to dc converter.

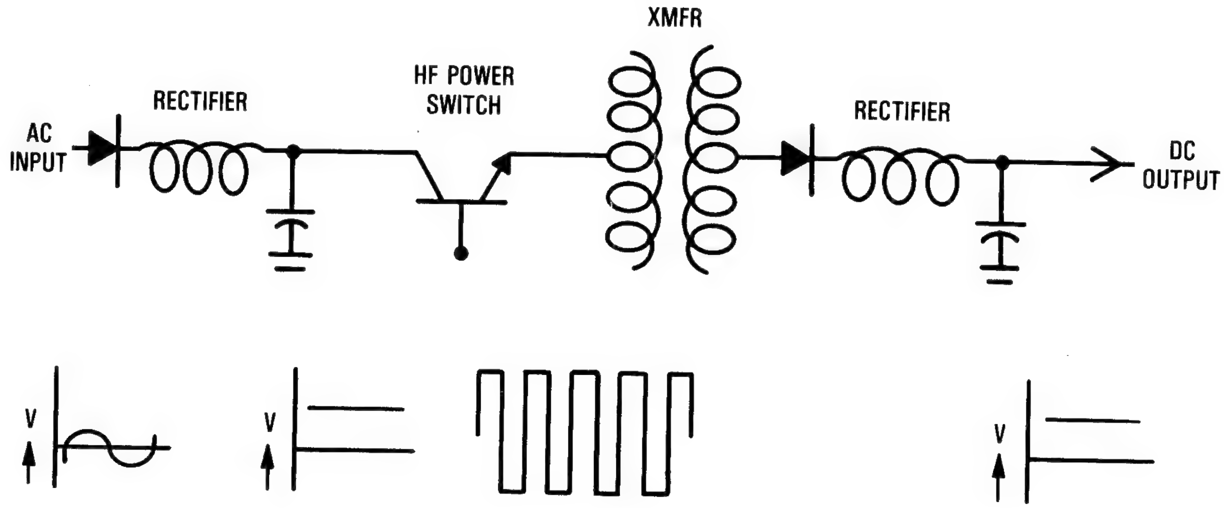


Figure 31. - Switched mode ac to dc converter.

Within the generic definition of switched mode inverters there are different types

- Fly back converters
- Direct forward converters
- Buck/boost
- Others

The inverters/converters have in common an objective to use high frequency switching from 2 to 20 kHz and higher. This requires low loss transformer/filter cores, such as Metglass, Permendur, and vanadium/Permendur.

- PWM (pulsewidth modulated) power supplies where a high frequency unmodulated waveform is pulse-width modulated.

  A special form of PWM is the programmable type for motor speed control, where "ratio-changing" is necessary using asynchronous or synchronous modulation techniques to vary the input/output frequency ratios.

- Cuk converters. - This is a special form of dc to dc converter and has the attributes that the input and output currents are nonpulsating. It is a step-up/step-down type that is equivalent to a buck/boost converter, see figures 32 and 33.

  Special integrated (coupled) Ćuk inverters not only are non-pulsating, but also have zero ripple on one side or the other.

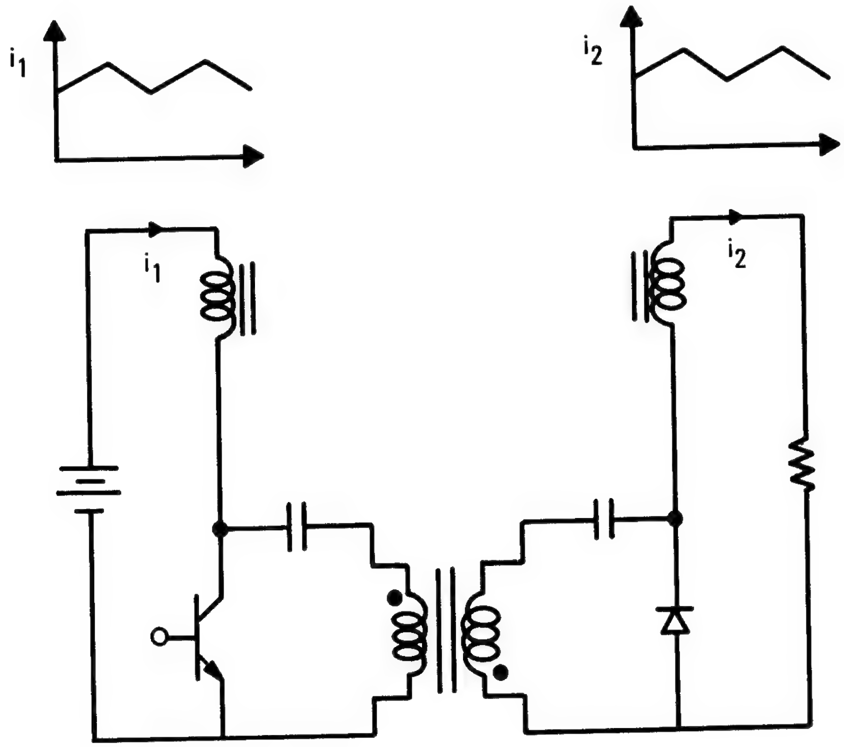
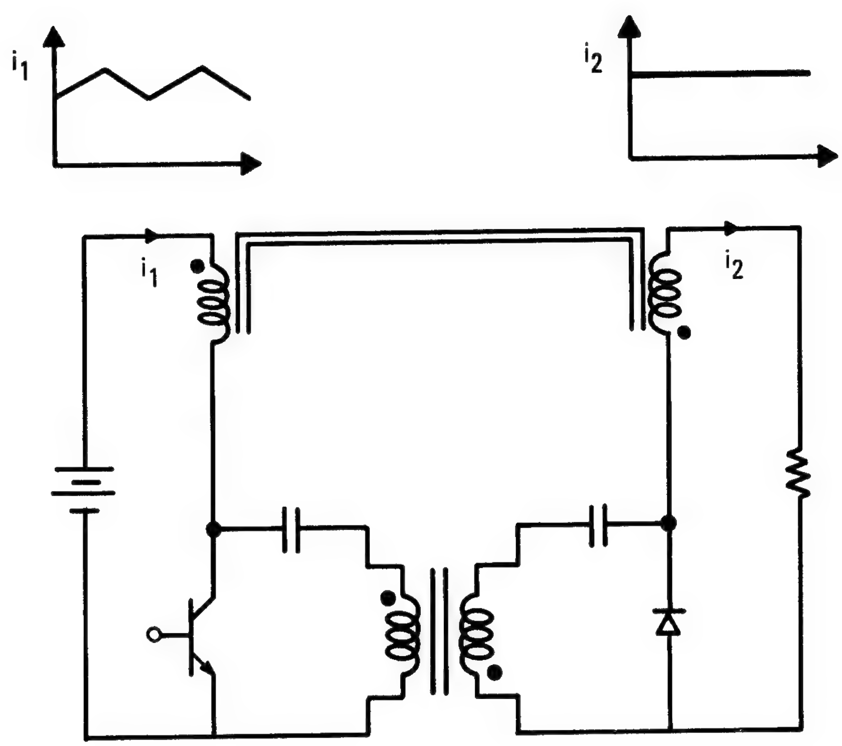


Figure 32a. - Cúk converter: nonpulsating currents.



Fibure 32b. - Cúk converter: magnetic coupled design.

- Resonant Mode Converters: these are the converters of specific interest (to the 20 Hz power system) and they are "four-quadrant" type that have the ability to be bidirectional and take ac or dc at either end.

The main advantage of the resonant power mode converter over other types is the four-quadrant universal type operation and its adaptability to large power capacities. The ability to utilize the resonant power converter technology in the large multikilowatt sizes is attributable to the low switching and parasitic resistance losses. This again is a result of the high speed switching characteristic that operates at the fundamental resonant frequency, and the fact that such switching is accomplished at zero cross-over. The high frequency sine wave characteristic and the zero cross over feature therefore lead to a converter design that projects low switching losses and high efficiency.

The choice of the solid state switching device is itself key to the converter design, so the development of bipolar devices, SCRs gate turn offs, (GTOs), gate assisted turn off thyristors (GATTs) and other advanced solid state switches pace the technology. The objectives are high voltage/current capacity and fast-switching (fast turn-off) times. Thyristor devices such as SCRs have relatively slow turn off time (vis a vis 10 to 50 kHz commutation rates), since it is necessary to remove the holding current and to reverse bias the junction. On the other hand, thyristors developed for line commutated inverters (such as AEG-Telefunkens TX 1050) can be rated for 3000 to 4000 A, rms, and for voltages up to 4000 V. Bipolar devices such as transistors offer much simpler turn-off control but are more limited in power capacity, having peak current capacities of 200 to 500 A. For large capacities, therefore, paralleled transistors are required. Gate assisted turn-off thyristors based on recent technology improvements have much shorter turn-off times and faster recovery times. These advanced thyristor types seem to be a logical candidate for the 20 kHz resonant mode converters. Gate turn-off devices and special bipolars are other candidates for future consideration.

Overall, as regards performance, the resonant power mode converter appears to offer advantages particularly in the large power levels over the pulse width modulation (PWM), pulse width amplitude modulation (PWAM), buck, boost, buck/boost and other inverter converter configurations. These latter conversion/inversion systems operate with relatively high efficiency, but have higher switching losses. Also, high frequency and choppers are manifest by switching noise and the need for effective input/output filters. The generation of wide range harmonics and concern for RFI suppression are therefore key factors in the trade analysis of converter/inverter technology.

Practical models of the resonant mode power converters have been built and work continues in evaluating them. The issue for the IDEA/Alternate IDEA is how the resonant power mode converter fits into the environment of an aircraft electric power distribution system and what are the installation considerations. In this regard, the first aspect is the power distribution wiring itself. Here there is the problem already alluded to of "skin-effect" and cable impedance effect. Both of these factors work to exacerbate the problems

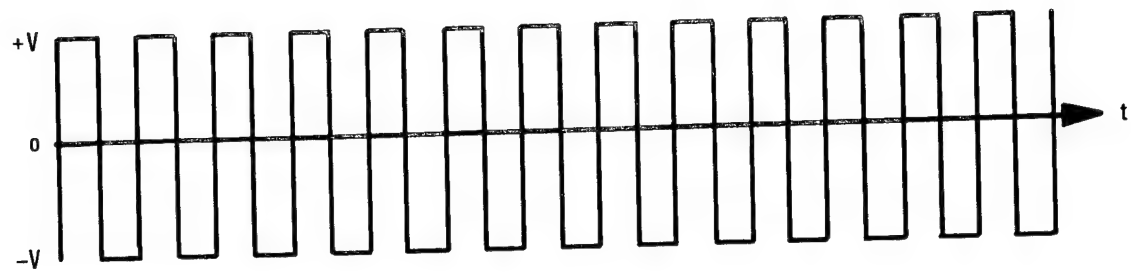


Figure 33a. - Ćuk converter: unmodulated carrier waveform.

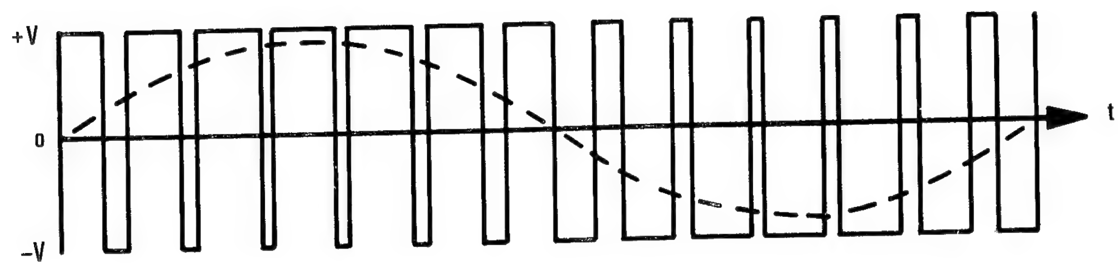


Figure 33b. - Ćuk converter: pulsewidth modulated waveform.

of long distributed feeders carrying ac currents. They are present with the 400 Hz and 800 Hz systems and, the larger cables reflect low power factors and higher voltage drop, due to the frequency-dependent ($2\pi fL$) effect. Also, the curves (Figure 57) show that at 3.5 kHz the impedance of an 0 gage cable is 6.4 times larger than is impedance at 400 Hz.

The result of skin effect and higher impedance voltage drops necessitate a new approach to hf cables in aircraft. Small stranded conductors insulated and fabricated with a special lay need to be developed. Small diameter wire could also be laid around a flexible plastic core or tube, using a bifilar winding technique to inhibit self inductance effects. Practical methods of splicing and terminating the special cables must be developed.

As regards the resonant power switching inverters, there appear to be potential advantages that would merit their consideration and use in the IDEA/Alternate IDEA aircraft. However, it appears that these advantages may be best exploited by distributing the power at high voltage dc levels, or at lower frequencies, such as 400, 800, or 1200 Hz ac. The disadvantage of the dc transmission system is not unlike that of the 20 kHz system. For example, dc is the most efficient supply source of long distance power transmission. However, in aircraft high voltage dc power can be used directly for only very few loads and extensive power conditioning must be employed for the many motors that operate basically simple modes. The 20 kHz is in much the same category as 270 Vdc; it is not as suitable for heating (galley), deicing loads, etc. The resonant mode matching inverters could, however, be interfaced with advantage to the conventional and semiconventional ac systems by using front end rectification and then converting to 20 kHz, or other suitable

resonant-frequency level. These inverters/converters could therefore be utilized in the IDEA/Alternate IDEA on a dedicated basis.

As previously highlighted, there has been ample theoretical assessment of the efficacy of resonant mode power conversion but it lacks the more extensive practical background experience resident in cycloconversion, dc link, and the various other high frequency switching mode type inverters/converters. There is also a plethora of solid state inverters/converters/ being used in industry for ac and dc motor drives from the sophisticated low power levels up to very high power levels. For example, drives are available for synchronous motors from 500 hp up to 30,000 hp. Adjustable frequency ac drives for induction motors and synchronous motors are highly developed in industry at this time.

From a technical standpoint, further laboratory work and testing is necessary to validate the resonant power mode converter for application to dedicated single functions or as a primary power source for multiple function/loads. In the latter case, the variable nature of the loads, in a typical aircraft, could present some nuances of operation that have not been presently addressed. For example, many previous solid state power inverters had very low source impedance with the result that, in the presence of heavy overloads and short-circuits, the "through-fault" current was so high as to cause "self-destruct". In the case of the dedicated resonant power mode converters they can be "tuned" to the load, but when they are used as a primary source the resonant frequency might become "detuned" by the nonlinear loads that draw harmonic currents. Also, the starting currents of large induction motors is a significant consideration: the low standstill-reactance of these motors result in short-circuit type currents at power factors that are very low during starting. When running, the motors present a high impedance and higher power factor load.

The series (current) and parallel (voltage) type resonant power converters could have different operational characteristics when used in the primary power source role. These should be fully evaluated in the ensuing development. The current type converter where the load is in series with the reactance and capacitance operates in a mode where the source-voltage is developed almost completely across the load, and is, therefore, a function of it. In the resonant mode, the reactive and capacitative voltages are antiphasal and, therefore, cancel each other so the source voltage is absorbed by the load. Should, however, a short-circuit occur, then there is no load resistance, and as in typical "acceptor-resonant" circuit, the current can reach very high values and high voltages will be developed across the reactors and capacitances. Similarly, while either type can be designed to resonate at the fundamental frequency, the presence of harmonic current distortion could present a problem to the tuned resonant mode.

The emergence of the 20 kHz resonant power converter technology becomes another interesting candidate in the proliferating field of high frequency switching mode inverters and converters. The 20 kHz resonant power conversion technology has potentially attractive features that need to be evaluated further. Of particular import is its efficacy as a primary power source because the aircraft application imposes a wide range of loads and loads that are non-linear in terms of their harmonic current draw.

Resonant power mode converters could still find application as dedicated power supplies for equipment and motors in the Alternate IDEA. The subject of static power inversion/conversion is an esoteric one and it is one that requires highly circumspect evaluation. Design considerations are sometimes antithetical: the requirement for high efficiency connotes low resistance, but low resistance connotes low damping, so the loop-dynamics and band-width/stability are affected. These are the design nuances that must be addressed in any closed-loop dynamic system, but they are key also to the achievement of stability and high performance in the static power inverters/converter technology.

3.2.8 Propulsion system interface. - The specific difference between the engines in IDEA aircraft (compared to the Baseline) is that the engine bleed air requirement is deleted and no pneumatic air is available for engine starting, air turbine motors, thrust-reversing, and hot air ice protection systems. This nonbleed engine therefore gains by a reduction in the SFC and thrust penalties on incident on the baseline airplane. There are also weight gains that derive from the elimination of bleed port hardware, bleed air ducting precooler installation, pneumatic starter installation, miscellaneous other hardware (such as splitter assemblies) and reduction in the engine weight. The E^3 technology used is based on a bleed configured design, so it is not cycle-optimized for shaft power extraction. In the absence of a specific cycle-optimized design, the physical differences in the IDEA engines are computed parametrically from the baseline E^3 design. The weight changes, when quantified, were cycled through the ASSET program.

The mission profile is based on a constant mach number and is controlled by the automatic flight management system (AFMS) for maximum fuel economy (e.g., max still-air-range/lb. of fuel). In this mode, the power settings on the engine are decreased on route as a function of altitude-increases and the reducing fuel load. In addition to the minimum fuel burn, the AFMS can be programmed for maximum range, shortest mission time, and other modes. The mission electrical load profile used for performance calculations is shown in figure 34.

The accessory gearbox (AGB) in the IDEA (as in the Baseline) is driven by the power take-off (PTO) shaft taken from a bevel gear drive on the HP spool. With the removal of the hydraulic pumps, the pneumatic starter, the CSD/generator assemby, the gearbox design, its weight and complexity are reduced. Consequently, it is referred to an "austere" gearbox. Functionally, aside from the reduction in complexity, the horsepower to drive this gearbox is proportionally much less because of the simpler gear train and the lower friction/lubrication oil/hydrualic pump losses. Also, the HP spool is relieved of high breakaway torque of the CSD system and the oil viscosity changes at low temperature during engine starting at -59°F. Starting of the engines in the IDEA is therefore improved during cold day operations since there is reduced viscous drag.

Another difference in the AGB will be a change to the torque/speed characteristics of the drive pads. In the Baseline, hydraulic pump speeds are approximately 1800 to 3800 rpm and the IDG AND 20006 pad for the 90/120 kVA IDGS about 4500 to 9200 rpm. In the IDEA, the pads for the two 150/175 kVA

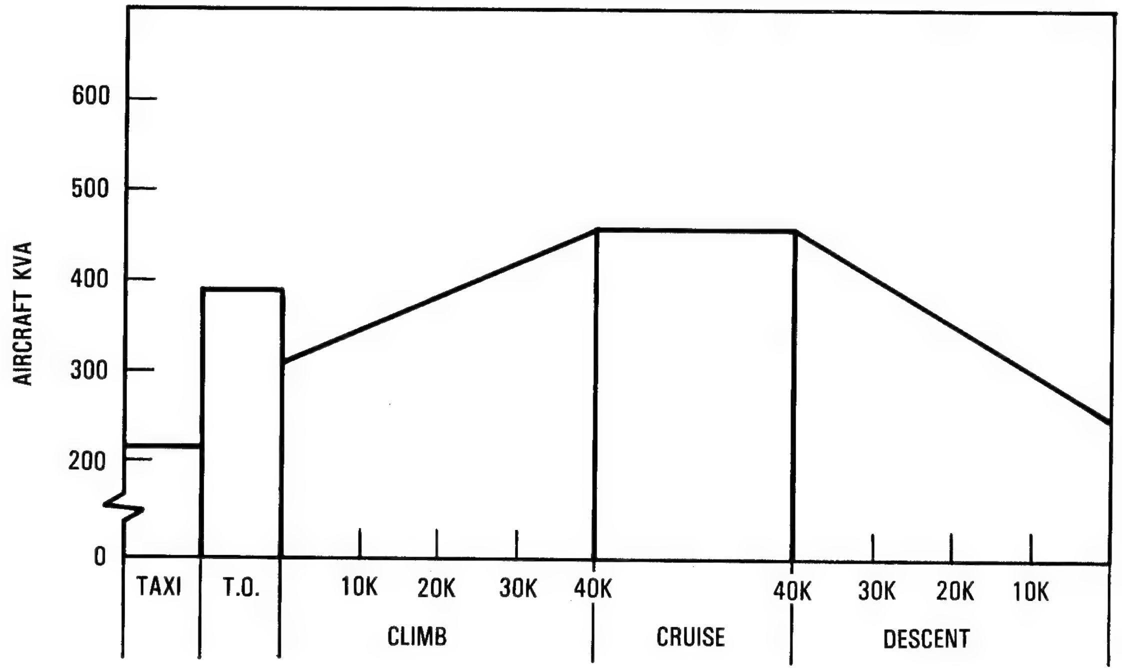


Figure 34. - Mission electrical load profile.

generators have an approximate speed range of 8600 to 17400 rpm. A number of drive configurations are possible in the IDEA that include fan case or core case mounted accessories. In figure 35 a "saddle mount" configuration is shown in which the two generators are mounted on diametrically opposite sides of the gearbox. This arrangement is further exemplified by the fact that the common drive shaft in the AGB is terminated at each end with "master" spline that mates with a similar male spline on the generator drive shaft. This ensures that the generator rotors are automatically "synchrophased", when installed. It also permits parallel operation of the generators in the engine-start or other modes. A similar concept, but using side-by-side generators, is shown in figure 36.

Figure 37 is another installation possibility that offers novel features. Here an in-line tandem generator is shown in which the mounting flange is picked up near its center of gravity. The two electrical terminal blocks are located at one end of the machine allowing the tandem machine to be inserted through the central mounting flange to connect into the drive geabox that could be integral or separate. The in-line configuration eliminates the need for mechanical synchrophasing since the machine designer can accurately control the angular relationship of the magnet poles.

The pylon mount illustrated in figure 37 uses a novel tower-shaft disconnect which is incorporated in the compact planetary gear configuration located on the top of the engine. The reaction torque of the ring gear is effected via a pin that inserts through a hole in the outer periphery of the ring gear. This pin can be removed from mechanical engagement by an electromagnetic solenoid that would be energized with the onset of any electrical shorts in the stator of the SmCo generators. Retraction of the solenoid pin

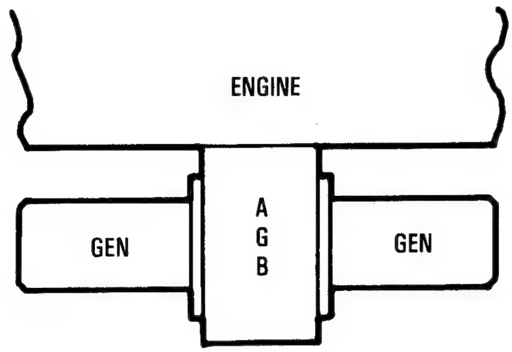


Figure 35. - AGB: generator saddle mount.

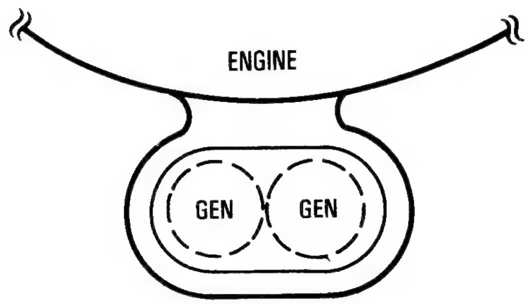


Figure 36. - AGB: side-by-side configuration.

would in this event allow the planetary gears to spin free. The solenoid pin approach is typical only since other electromechanical actuation is possible.

Other refinements (shown in the inset diagram) show the use of an electro-optic alignment method to permit reinsertion of the pin on the ground. A power drive method (power drill) allows for rotation of the ring gear assembly until the pin is lined up with the hole. A photo-transistor senses this condition via the incident light from a light emitting diode (LED) focused on the reflector located on the outside of the ring gear. An advantage of the pylon installation is that it permits a very clean engine installation, but accessibility (for maintenance, etc.) is reduced. The inline (tandem generator) configuration is otherwise applicable to a conventional AGB mount configuration.

The engine supplier will be requested to integrate the AGB oil lubrication and oil cooling provisions into the engine system. As described in the trade analysis of alternative electric power generator systems, it is also strongly recommended that in the IDEA the cooling of the SmCo generators be integrated into the engine oil cooling system so as to avoid additional complexity of mechanically driven pressure and scavenge pumps, and filters. A spiral housing may be used to enclose the back iron of the generators and fuel, oil

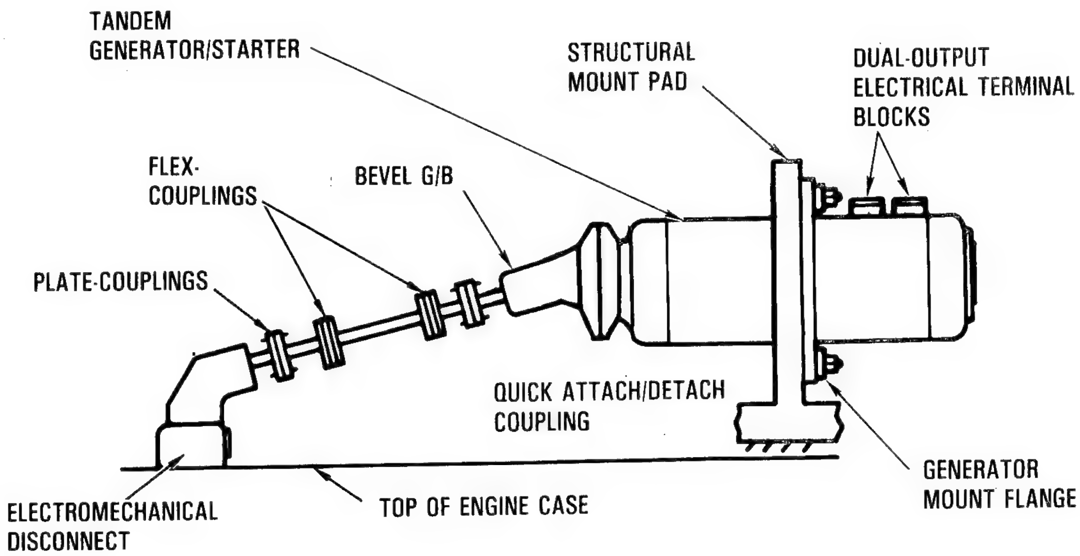


Figure 37a. - Tandem generator: typical installation.

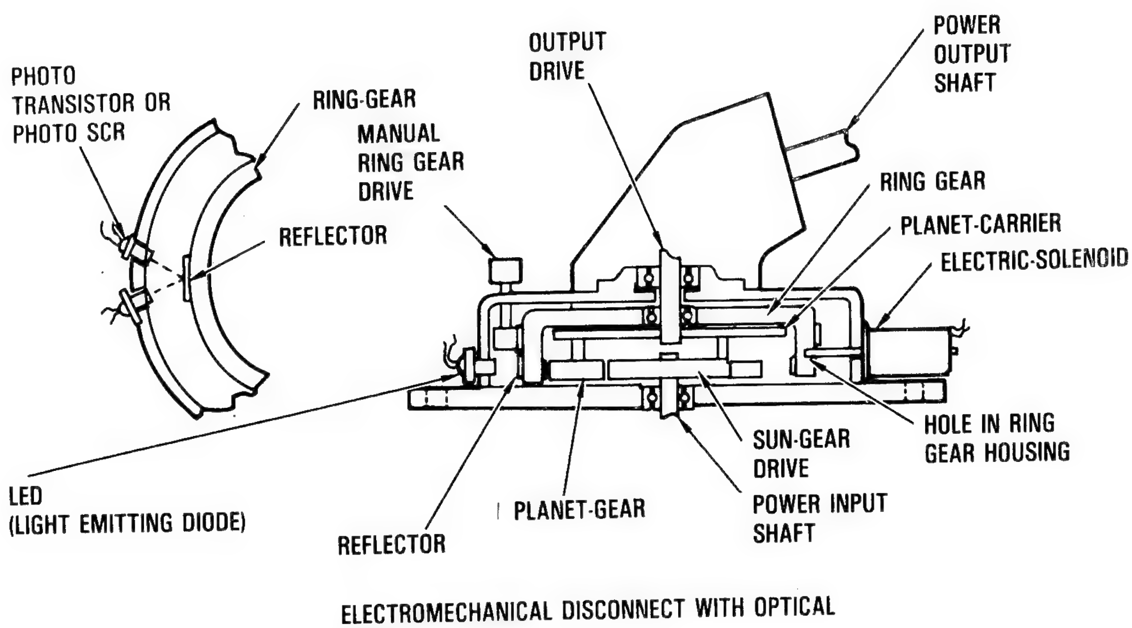


Figure 37b. - AGB pylon mount: in-line tandem generator with power shaft disconnect.

or other coolant could be pumped through this highly reliable heat exchanger circuit. There would be no contact of the fluid medium with any of the internal electrical parts of the machine. Because of this highly reliable cooling method for the tandem generators, it is not expected that the engine supplier would be concerned with the approach of integrating the cooling in with the engines oil cooling system.

3.2.9 Engine starting. - This function in the IDEA is furnished by operating the SmCo generators as synchronous motors during the start mode. In the VSCF type systems the static power converter/inverter (in the output lines of each generator) is used in a "reverse power mode" and the voltage/frequency applied to the stator of each starter-generator (S-G) is programmed to control the acceleration rate and start-time of the engine. A prerequisite of this operation is that the commutation of the SCRs or the power transistors in the power electronic assembly be controlled by a rotor position sensor incorporated in each S-G. In the direct drive generator system (DDGS), or variable voltage/variable frequency (VV/VF) system, a separate/dedicated inverter assembly is used to generate a quasi-square wave output for the S-Gs. Two starting inverters (to provide redundancy) were designed by Sundstrand Aviation for the function and each weighs approximately 55 pounds.

A typical engine start characteristic for an E^3-type engine is shown in figure 38 for P&W STF-631-type engine. The HP spool on this engine has a polar moment of some 9.6 slug ft and its -40°F peak drag torque at light-off speed of 2600 rpm is approximately 130 ft-lb: minimum self-support speed/ground idle speed of the engine is approximately 8600 rpm. Since the engine in the IDEA is the same as in the Baseline, the starting characteristic is the same except for the differences of higher breakaway torques, reflected by the accessories in the Baseline during cold start conditions. However, it can be seen that based on peak torque at light-off of 180 ft-lb, the horsepower delivered to the engine HP rotor is approximately 90 HP.

As stated, in the horsepower rating of each SmCo generator for starting the engines in the IDEA is not as high as on the RB-211 and CF6 engines, but torque is still a key sizing factor in the generator. Based on a HP spool speed of say 12,000 rpm, an approximate 1.33:1 step down ratio, from the generator to engine (16,000/12,000) is necessary. This gear reduction provides torque amplification but the generator's full load torque rating would be defined as follows:

$$T = \frac{5250 \times (kW \times 1.342) \times \eta}{N} \cdot K$$

$$= \frac{5250 \times 175 \times 1.342 \times 0.87}{16000} \cdot 1.33$$

$$= 90 \text{ ft-lb}$$

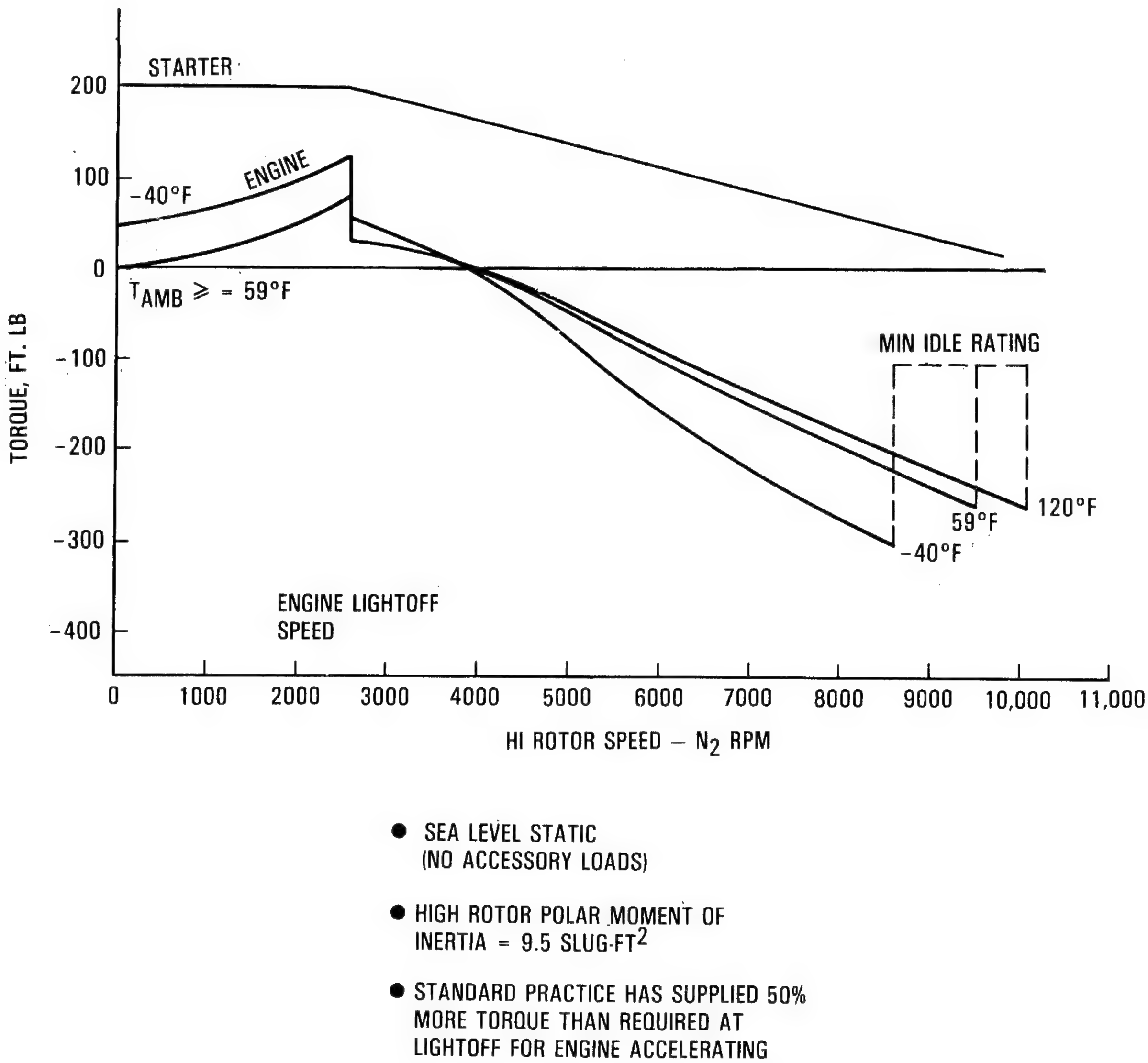


Figure 38. - NASA E^3 technology estimated starting torque.

Where T = torque in ft-lb

η = transmission efficiency

N = generator rpm

K = gear reduction ratio

This shows that to start the E^3 one of the two generators in each power plant, would necessitate going into the machine's overload rating. The alternative is to parallel the two generators, since mechanical-indexing of the rotors permits this. Clearly, the capability for using only one of the

two generators for starting would be an advantage and any increase in the electrothermal capacity of the S-G to do this would be worthwhile. The start mode of the electric machine nonetheless represents a more arduous condition than the generator mode. It is possible also to use an additional gear reduction in the start mode and bypass it in the run mode, but this adds the additional mechanical complexity of gears and overrunning clutches. It is also possible to consider some other completely different proprietary approaches to engine starting in the IDEA.

3.3 APGS Trade Analysis

The trade analysis and selection of the alternative power generation systems was difficult because some of the systems, while satisfactory for particular applications were not so suitable when applied to the higher capacity system as exemplified by the IDEA and Alternate IDEA. Also, it was clear that while system weights are usually important to the trade analysis, in the IDEA/Alternate IDEA, the DFCS control system (and the electromechanical actuator system for the primary secondary flight controls) were critically dependent upon the reliability of the APGS.

Safety is key in defining the reliability statistic of not more than one catastrophic failure in 10^9 hours or 10^{-9} failures/h. In the conventional aircraft such reliability/failure statistics apply mainly to the structural integrity of the aircraft but they also apply to the flight control system, regardless of its design configuration and its source of power. The advanced technology precepts of IDEA/Alternate IDEA, however, add a new dimension to the critical analysis of the safety aspects, because not only is there a completely new method of actuating the control surfaces, but the actuation system is highly dependent upon an ultra-sophisticated DFCS, which in turn is critically dependent upon the reliability of the APGS.

Sophisticated software, and its verification and validation further add to the overall concern for safety requiring that the same critical attention be given to the software design/engineering as to the design of the DFCS itself. Notwithstanding, the IDEA/Alternate IDEA is confidently committed not only to its operation in regimes of relaxed static stability, RSS, but also negative static stability in which pitch active control system (PACS) has to provide reliability statistics commensurate with the airplane's safety objectives.

Reliability/safety thus stand as the key concerns and transcend the other parameters of this trade analysis. Since reliability is related strongly to the simplicity of design, it can be seen that complexity and sophistication tend to abrogate the reliability criterion. Therefore, simplicity in the power generation concept becomes an important yardstick of evaluation. However, there is a caveat inasmuch as it is not a stand-alone statement, i.e., the simplicity is not exclusive to the design and hardware implementation of the power generating channel itself. For the APGS to score high marks, it must be simple in itself and it must not result in complications, increased weight or high maintenance support in the aircraft's systems and

utilization equipment. An example of this would be a requirement for excessive power conditioning in the utilization system.

Maintenance actions/maintenance support were noted above and this becomes the next important parameter of assessment. To a major extent, excessive maintenance support costs hark back to reliability (MTBFs, TBOs) and the complexity of the systems installation. Clearly, complicated/sophisticated systems require high standards of technical support on the ground, and the more involved the systems the more detailed the maintenance/checkout procedures and the more protracted the corrective actions. The downtime as a result of these actions decreases the availability of military aircraft and reduces the operational effectiveness of the airlines.

A corollary to the above relates to the mean-time-to-repair (MTTR). Here the figure of merit relates again to complexity, since the higher the parts count in a system or equipment, the lower is the statistical reliability. However, some equivocation is required because the supplier of advanced sophisticated equipment has moved more towards modularity in design and is using dedicated microprocessers to monitor the status/failure modes of the systems/equipment. Such dedicated microprocessors isolate failures down to the LRU level and permit ready removal of the LRU for further diagnostic evaluation in the maintenance shop. Excessive use of BITE and modularity in design should not, however, preempt or usurp the objective of achieving the required performance with the simplest and most reliable system/equipment.

The suitability of adapting the APGS to high power levels is another important weighting parameter. Some sophisticated, and otherwise highly suitable electric power systems, may not be suited to the IDEA/Alternate IDEA application since the total installed capacity requirement could be much too high for 100 percent power conditioning. In the advanced military transports major power demands may derive from the application of sophisticated military weapon systems.

"High power," in turn, emphasizes high overall transmission efficiency and the need for reducing the heat dissipation characteristic. It is clear, if the APGS demands major thermal management design consideration, it will impact unfavorably on weight, size, reliability and cost. Efficiency and thermal management therefore become important additional evaluation factors.

Weight is a consideration in any aerospace system and the industry is conditioned to addressing this aspect in the design and development of lightweight systems and equipment. A possible difference in the IDEA/ Alternate IDEA is that while weight is important, the reliability criterion is more important, and random failures should not be a legacy of weight reduction efforts.

In the trade analysis itself, a reasonable amount of effort was directed to a "critique" of the CSD, VSCF, and 270 Vdc systems because they are technologically mature and competitive systems within their own right. In the case of the VV/VF and the 20 kHz systems, the treatment was somewhat different since these systems are immature and without the benefit of significant background. Consequently, these systems were not critiqued, per se, but

instead they were technologically evaluated more in a tutorial manner. This was done because there is a lack of understanding as to practical implementation aspects of these systems and there is a concern in the industry as to their total efficacy when applied to future transport aircraft. As regards the differences between the VV/VF and the 20 kHz power systems, the latter is more in need of investigative analysis and design activity by the aerospace electrical industry since it is not possible at this time to quantify the performance, size, weight, cost, reliability and other details, in a real world implementation of the system. The same is not true of VV/VF since it takes advantage of many of the technologies that now reside in the aerospace industry, e.g., samarium cobalt generators/motors, induction motors/generators, solid state power switching and solid state power conversion/inversion technology. At the outset, the VV/VF should enjoy the attributes of simplicity, reliability and high power transmission efficiency.

3.4 Selected Power Generation System

The decision and selection of the electric power system was difficult because of the mature status of many of the competing systems for the IDEA application. While a matrix comparison with numeric ratings assigned for the respective evaluation parameters could have been used, it would define the selection in an unequivocal manner and it would mask the benefits other systems might have were some of the ground rules different. For example, the selection process could have been influenced in a different manner if the capacity of the individual generating channel were say 90 kVA or less. Also, the number of motor loads in an airplane have a major bearing on the design configuration and efficacy of a particular system. Clearly, if power is generated at a voltage and/or frequency level that necessitates excessive power conditioning, then the total system suffers in terms of weight, complexity, reliability, maintenance support and cost. In another airplane, such as a military type, it is possible also that the weapons system and avionics could be key and have a major influence on the design of the APGS. For example, in military aircraft there is also a different interplay between the requirement for cooling and pressurization. In a large commercial airplane the pressurization requirement at altitude is the dominant load, compared to the cooling requirement, whereas in a large military ASW type aircraft, the pressurization requirement could be low and the cooling demand high.

Acknowledging the above design factors and the type of loads in the IDEA configuration, a variable speed variable frequency VSVF (VV/VF) type system was selected for this airplane, having two 3-phase 400V 800 Hz 150/200 kVA generators in each of the three power plants.

Figure 39 is a bar chart peak load analysis covering the different segments of the mission profile, showing that the maximum continuous load demands occur during the 35,000 ft/0.8M cruise condition. This is the condition where full cabin pressurization is effective and where other large loads such as the galley, flight controls, lighting, avionics and other loads such as fuel boost/transfer pumps, are in operation. Based on these loads, the maximum continuous rating of the tandem generators in each power plant is approximately 160 kVA or 80 kVA/generator. As indicated, the cabin

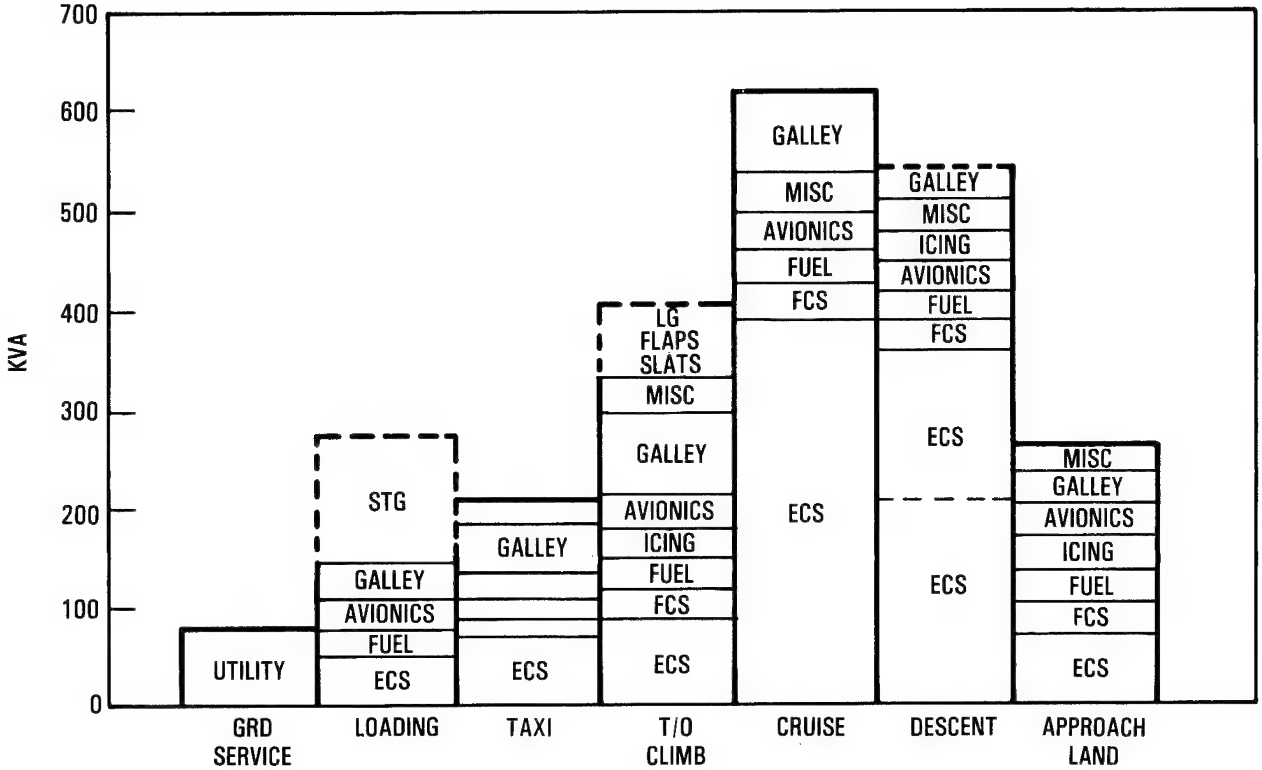


Figure 39. - IDEA airplane peak electric loads vs mission segment.

pressurization requirement is the major load in the airplane and unlike the galley, it reflects a continuous power demand to the electric power system.

Using a nonregenerative ECS approximately 140 hp would be required to supply 1.4 lb/sec of bleed air per engine. For performance calculations in this study it was assumed that a regenerative ECS be used in which ECS air, instead of being dumped overboard, is passed through a heat exchanger and then through a power turbine attached to the ECS compressor, thus decreasing the ECS electrical power requirement. In this manner cruise ECS power is reduced to 94 hp/engine. As shown in Chapter 11, non-ECS loads for the Baseline and IDEA are very similar, since those functions that were performed hydraulically on the Baseline are performed electrically on the IDEA.

The conclusions and summary of the above overview are: -

(1) The choice of the APGS in the IDEA is dictated, more than anything else, by the magnitude of the loads and the type of loads in a large commercial international range transport.

(2) The horsepower demands of the "utility/FCS" loads, are approximately the same, in the baseline and the IDEA.

(3) The major difference is in the comparison of engine bleed vs. shaft power extraction as qualified by the above discussion.

Finally, it is relevant, as described later, that the motor driven cabin compressors and the ECS in general are powered directly from the engine driven generators, without intervening electronics.

3.4.1 Electric power system design. - The primary power supply for the IDEA comprises six-engine driven SmCo generators (two per power plant) operating in a nonparalleled configuration, and each having a nameplate rating of 150/175 kVA. Because of the high power level and to save cable weight and electromagnetic weight, a double standard voltage and frequency system has been selected. This means that the 6-pole generators develop 3-phase 400 V 800 Hz power at a nominal speed of 16,000 rpm which corresponds approximately to 92 percent high-pressure spool speed. Each generator, being a permanent magnet (Sm_2Co_5) type machine, develops voltage that is approximately proportional to engine speed, so the following electric power characteristics prevail over typical engine speed conditions:

Condition	Engine Speed	Volts	Freq (Hz)
Take-off	100%	435	870
Max Cruise Climb	96%	418	836
Altitude Cruise	94%	408	816
	90%	392	784
High Altitude Descent	85%	370	740
Approach	80%	348	696
Ground	64%	278	556

As evident, the E/F ratio remains approximately constant over the total speed range and it is a characteristic that is ideal for ac induction motors. The primary ac voltage is also suitable for most heating/lighting loads, transformers and other electromagnetic equipment. During the cruise regime and practially all other parts of the mission, the engine speed variations are small, so an approximate constant voltage/constant frequency characteristic prevails over most flight conditions.

While the primary power system operates as a variable voltage/variable frequency (VV/VF) system, it is projected in the time frame of the IDEA that there will be a significant inventory of equipment still requiring the conventional 3-phase 200 V 400 Hz type power system. Because of this, four 25 kW constant voltage/constant frequency (CV/CF) electronic assemblies furnish the demands for high quality CV/CF power in the IDEA. Other special power supplies, such as 270 Vdc are derived from 3-phase phase delayed rectifier (PDRs) assemblies that front-end the power electronics of the FCS servo actuators. Four dedicated PDRs could have been used to furnish 270 Vdc, but the distributed approach was selected. Finally, 28 Vdc power can be supplied either

by unregulated TRUs (transformer rectifier units), or by switching regulator power supplies powered from the dedicated CV/CF power system. The latter could also be used for other special low voltages 5, 12, 15 V power supplies. The dedicated power supplies are discussed further in this Section.

3.4.2 Generator power system control. - Each power channel incorporates a sophisticated supervisory control that furnishes the necessary logic for the control and protection elements for the complete power system. As with the baseline systems, a microprocessor-based supervisory panel is used for each power channel and it operates autonomously, outside the digital load/power management system. This latter system (described in Section 5) is responsible for "total system" management and it will not interfere with the role of the microprocessor based supervisory panels in their real-time monitoring of the individual power channels. Any anomalies in the system voltage/frequency, presence of power feeder faults (open circuit, open phase, short circuit, differential fault) are policed by the individual supervisory panels and appropriate action is initiated to avoid problem conditions. Abnormal system behavior, loss of engine/generator, etc., are also monitored by the supervisory panel and information is flagged to the central digital power/load management system, which will take appropriate action to reinstate any power loss to particular electrical section. The central digital processor system also incorporates the logic to effect an orderly and sequential disconnection of loads, to avoid any overloading of remaining generators or individual power feeder sections.

To a major extent, the microprocessor based supervisory panel in each power channel is simplified to the extent that it is not necessary, as in a paralleled electric system, to monitor and control the real and reactive power balance between the paralleled, electric generators. The system is also free of synchrophasing procedures, necessary to effect paralleling, and the sophistication of providing a real-time unambiguous isolation of aberrant power channels. Similarly, the generators are direct engine driven, so there can be no overfrequency/underfrequency, overvoltage/undervoltage conditions, unless the engine itself overspeeds or underspeeds. Clearly, any speed anomalies in the engine will be monitored closely by FADEC (Full Authority Digital Engine Control) and appropriate action will be initiated to correct the anomaly by the engine monitor system or shut down the engine.

3.4.3 Electric generator design. - A desired characteristic for the generator in the IDEA is for it to have a low voltage regulation. The extent to which this is realizable will be subject to a detailed generator design, but a major simplification could accrue to the power system if the generators possessed inherently stiff voltage characteristics. To accomplish this, the generator would have to be designed with a larger than normal air gap and with magnets that are relatively insensitive to strong demagnetizing fields B-H values). The development of the rare earth permanent magnets with increasingly high energy products and a flatter B-H characteristic with no sharp knee will contribute favorably to the design of such an inherently stiff electric machine. Samarium cobalt magnets or neodynium-iron-boron (Neomax) will offer energy products of over 35 x 10^6 gauss-oersted. Other new magnet materials will evolve with even higher energy products and more stable B-H characteristics. It is further possible, with different iron-to-copper ratios, to

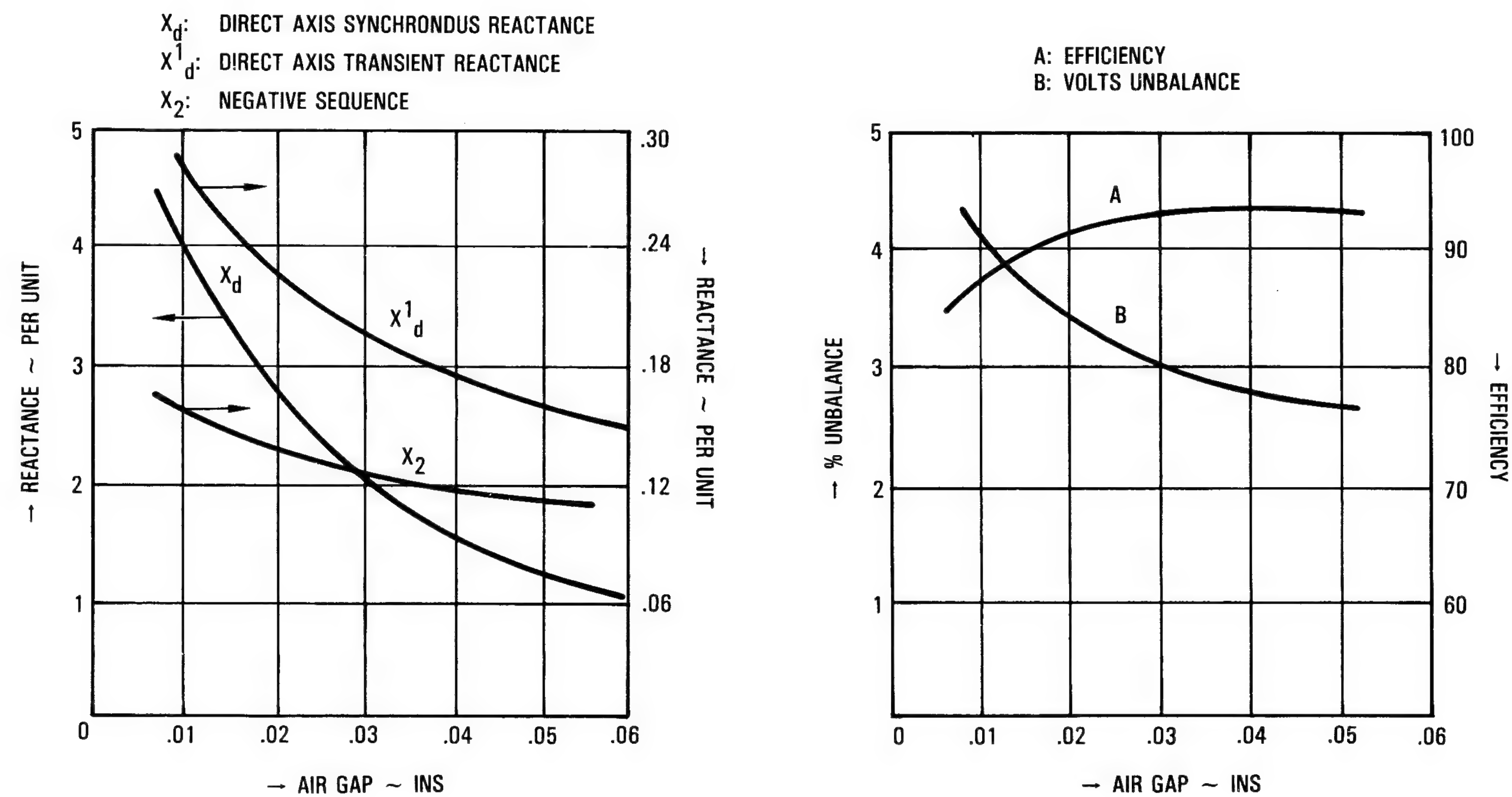
control the reactance values of the machine so that the generator is less sensitive to load changes and load unbalance. The direct axis reactance value, X_d, is itself strongly influenced by air gap length and decreases with increase in the air gap length.

Figure 40 is illustrative of how the generator's reactances vary as a function of the air gap length. In many current aircraft generators, the direct axis synchronous reactance is typically about 2.5 to 3.0 per unit. An X_d of about 1.5 PU however would be a "stiffer" machine from the standpoint of voltage regulation and consequently less sensitive to load changes. The X_d is of interest since this also decreases with increase of air gap, so again it is a favorable trend, in that the transient overvoltage and undervoltage on release and application of load are reduced. Similarly, the X_2 negative sequence reactance reduces favorably with air gap length and this reduces the voltage unbalance with uneven loading of the three phases. This latter effect is shown in figure 40b, where it can be seen that a machine with an X_d of 1.5, and an X_2 of 0.12, would have a voltage unbalance (with MIL-G-6099 specified unbalanced loading) of about 2.7 percent, compared to about 3.2 percent with a conventional machine. Finally, it can be seen that the generator efficiency characteristic could be maximum at the larger air gap values.

Electric machine design is therefore key to the performance of the large capacity generators in the IDEA and Alternate IDEA and the machine designer has the ability to evaluate many computer designs as he changes the different electrical and mechanical parameters. Presently, most aircraft generators are designed for minimum weight and performance compliance with prevailing aerospace electric generator specifications such as MIL-G-6099 and MIL-G-21480. Historically, the machines have small air gaps and high synchronous reactance values. As a consequence, they have inherently poor regulation and lower short circuit currents. The computer design that calls for large air gap and low voltage regulation will, on the other hand, be less sensitive to load changes and will correspondingly exhibit high short circuit current characteristics.

Wire gage is another influential factor in the generator design since high electric loadings (high amps/in^2) will not only increase the I^2R losses and decrease the efficiency but will also increase the IR drop component. Very high current loadings will also increase the conductor and insulation temperature and this too will increase the wire resistance and so increase the IR drop. The design objectives are to keep the IR and IX voltage drops in the generator low and to minimize the effect of armature reaction or the demagnetizing cross ampere turns.

In the evaluation of alternatives to low regulation machines, it is also possible to consider the use of electro-dynamic control methods that will not only provide regulation of the generator voltage, but also mute the permanent magnet excitation to avoid the necessity of mechanical isolation of the machine when stator shorts occur. Finally, there is the simple artifice of establishing the per unit regulation point of the machine, at a typical quiescent load level as may be typical of the cruise altitude operating condition of the generator. In the IDEA the ECS load reflects a relatively



(a) REACTANCES VS AIR GAP

(b) VOLTS-UNBALANCE, EFFICIENCY } VS AIR GAP

Figure 40. - Generator design characteristics.

high steady state demand and so it would be possible to establish the per unit air gap flux in the machine at this load level thereby limiting the regulation range.

3.5 IDEA Electric Power System Configuration

The electric system in the IDEA comprises the following:

- Primary ac power system
- Special dedicated power supplies
- Flight crucial power system
- Power distribution system
- APU/external power.

Figure 41 is a simplified electrical schematic of the primary ac power system that consists of six three-phase 400V 800 Hz 150/175 kVA direct driven samarium cobalt, generators, operating in a nonparalleled configuration. As previously described, each of the six channels incorporates a microprocessor based supervisory panel that monitors performance status of the generator channels and interfaces their control with the digital power/load management (DP/LM) system (which exercises total system management). Engine/generator failures, fault diagnostics, BITE monitoring are all included in the supervisory panels with appropriate status information being flagged over the digitial data data bus system to the flight station. The supervisory panels operate autonomously to de-energize and trip faulty power channels, but the DP/DL system accepts flight station commands while its processer system effects control over the multiple power feeder section breaks to ensure power continuity when engines and/or generators fail.

Conventional protection is afforded each power channel and the following are incorporated:

- Over/undervoltage
- Over/underfrequency
- Negative sequence current
- Phase sequence.

As previously stated, it is possible that the OV/UV and OF/UF protection could be eliminated by taking advantage of the overspeed/underspeed control protection on the engine. The typical differential protection is not used in the IDEA since, in a distributed bus system, it is difficult to provide feeder protection zones. Therefore the negative sequence current is used and this will protect against any line-to-line and open phase faults in the distributed buses. A further recommended feature is that an insulated generator neutral

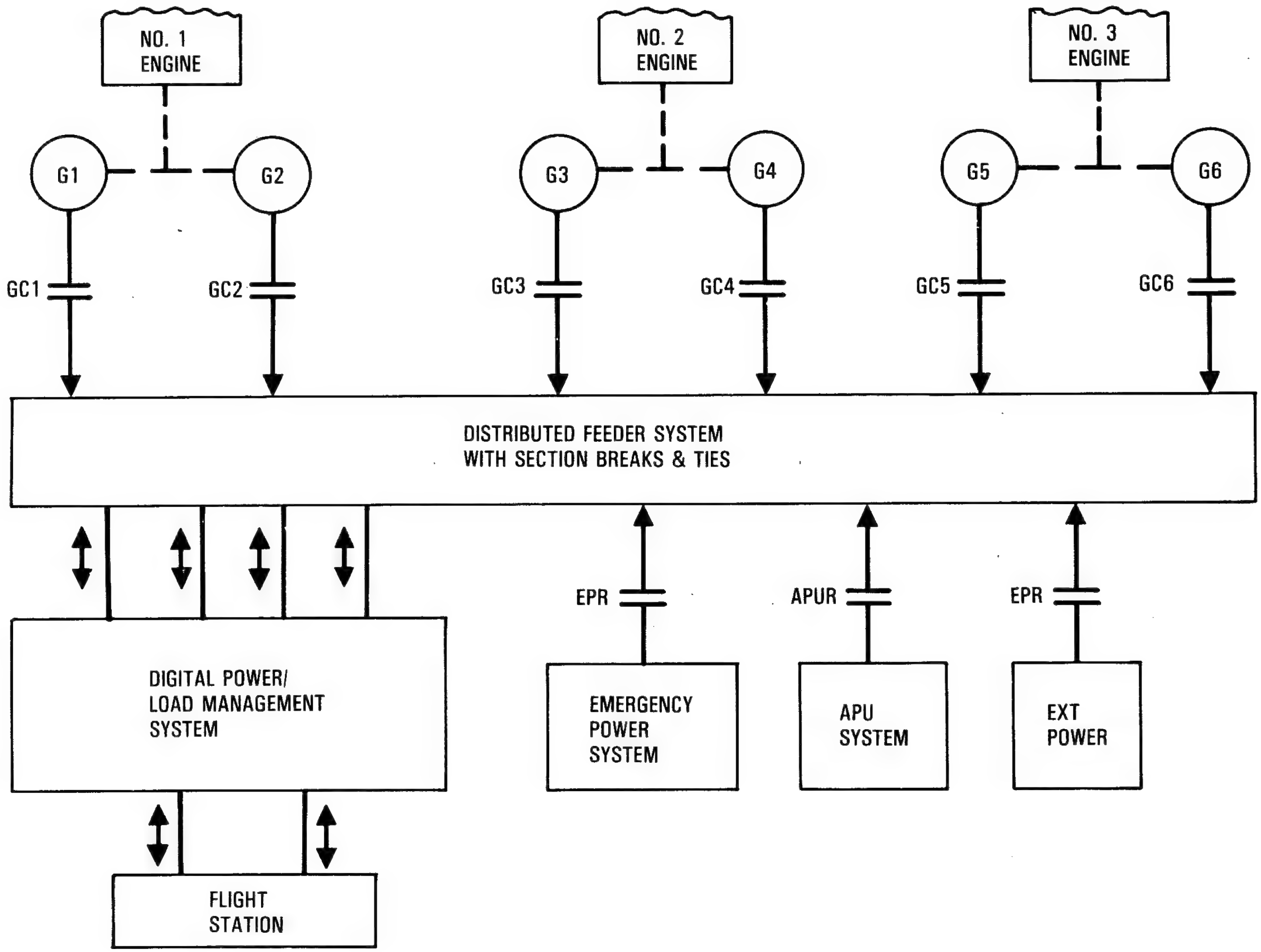


Figure 41. - IDEA primer electric power system configuration.

feeder or high impedance grounded neutral system be used to avoid the high destructive fault currents that are incident on line-to-ground faults.

3.5.1 <u>IDEA: digital power/load management</u>. - A primary function of the DP/LM system will be the management of the feeder section ties/breaks and the operation/interface of the aircraft's primary ac system with the emergency power system, the APU system and the external power system. An erasable/programmable read only memory (EPROM) is incorporated within the DP/LM system and provides for full automatic control of the complete electric system: it also accepts manual input commands from the flight station. Appropriate signal sources/transducers are incorporated to monitor the status of all contactors (section ties/breaks) and the operational status of external/APU power, etc. Responding to the automatic/manual inputs and the logic in memory the DP/LM system sequences the proper closure and opening of all contactors/bus section ties in response to operating conditions. Such control must have the highest reliability and integrity since the DP/LM system is responsible for ensuring power continuity and ensuring there is no inadvertent paralleling of the various electrical power sources.

A subset of the power management function incorporated in the DP/LM system is the equally-important load management system. This also is an integral important element of the DP/LM system since, while the power management system operates to transfer generators to different feeder sections the software program will act to effect a disconnection of loads in accordance to their assigned priority levels. The software program actually establishes the hierarchial level for all the loads in the IDEA/Alternate IDEA and this will determine their priorization status.

3.5.2 <u>IDEA: dedicated power supplies</u>. - These relate to those power supplies that furnish special power charateristics to selected systems and equipments. The 3-phase 200V 400 Hz CVCF system is one of these systems and it comprises four 25 kW static power electronic assemblies. These supplies provide the quad redundant power sources for the DFCS and also furnish power to the non-FCS avionics system and other loads that require CVCF power such as synchros, gyros, fans, T/R units.

Figure 42 is a schematic showing that the quad redudant 3-phase 200 V 400 Hz power supplies are powered from the quad redundant ac power feeders that are routed in the fuselage of the IDEA and Alternate IDEA. As shown, the front end of the static power supplies consist of a 3-phase SCR rectifier bridge to permit a 270 Vdc constant input voltage to the inverter section of each of the 25 kW power supplies. A tap is also made on the output side of the SCR bridge the purpose of which is described later. However, in normal operation, the SCR bridge acts as a phase delayed rectifier (PDR) assembly in which the phase angle of the SCRs are advanced and retarded in response to the voltage level of the primary ac generator system. Thus, during cruise flight with the three-phase ac line voltage at 400 V the SCRs are phase delayed, while at idle descent let down and on the ground the SCRs will be phase advanced. In this manner, 270 Vdc power is maintained constant over the approximate 2:1 voltage range of the primary ac power system.

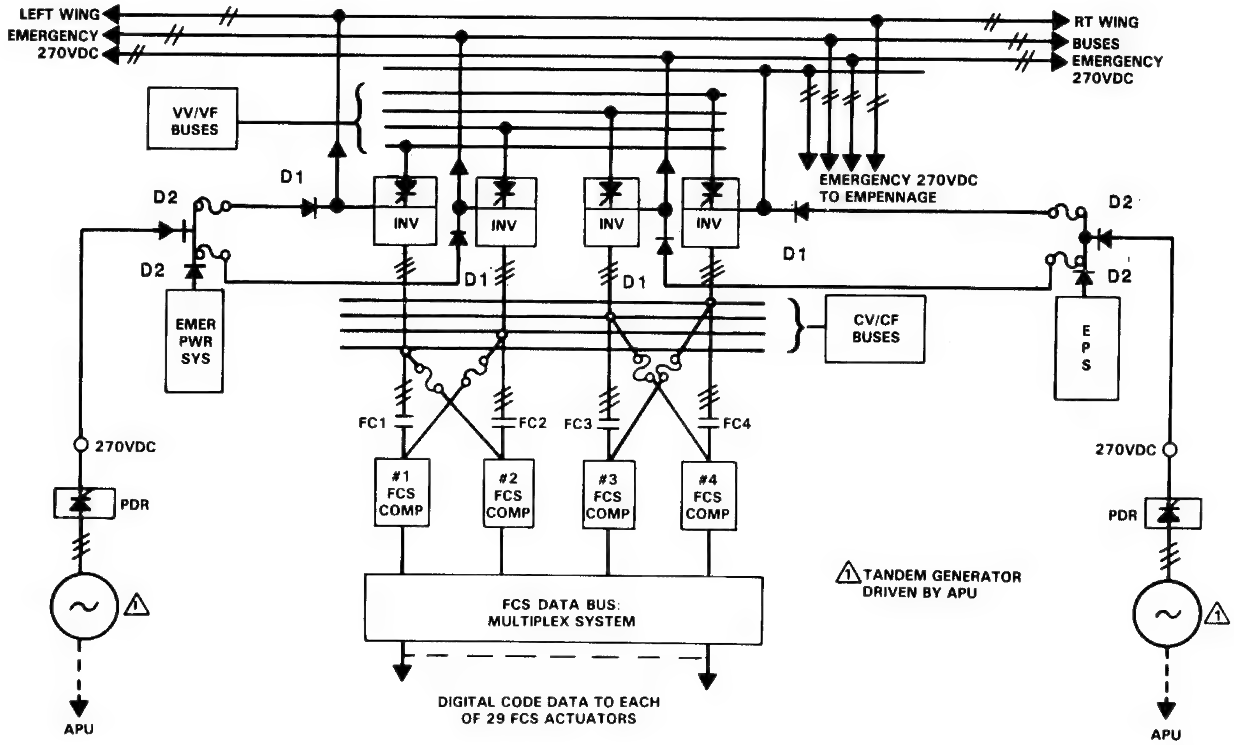


Figure 42. - IDEA flight critical electric power.

3.5.3 Flight crucial power system. - The emergency power system is key to the flight critical loads and to the achievement of the 10^{-9} reliability objective.

It is important to the reliability modeling and the overall complete design of the IDEA electric power system that the critical attention be paid to the avoidance of complexity in each power generating channel, to achieve the highest reliability and integrity. Similarly, specific attention must be given to the redundancy and reliability of the power feeder distribution system itself. The IDEA has six generator sources, the generators are non-paralleled, and the feeder distribution is designed to provide quad redundancy in the wings and fuselage. These feeders are physically/spatially isolated and they are insulated from ground potential. The primary power source reliability in the IDEA/Alternate IDEA is therefore very high and it is backed up by an ultra high reliability emergency power system that is dedicated to support the DFCS and other flight critical loads.

Figure 43 is a schematic of the flight crucial power system showing the multiple power sources available to the quad redundant inverters that furnish power to the DFCS computers. As shown, the four CVCF power supplies are

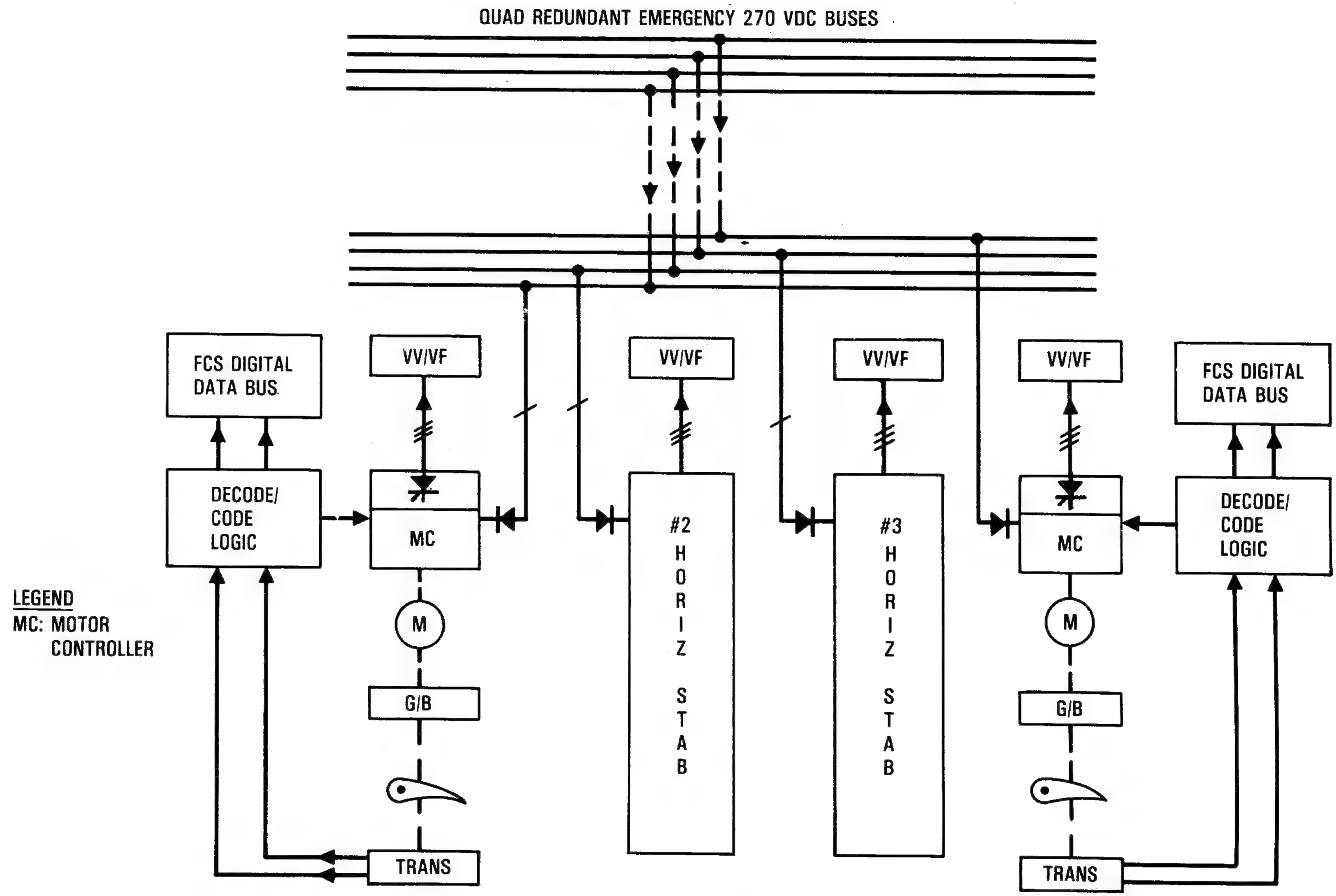


Figure 43. - IDEA flight critical emergency 270 Vdc power.

supplied by the quad redundant primary ac feeders. After rectification to 270 Vdc by the 3-phase SCR rectifier bridge that front ends each inverter a tap is provided to permit the inverters to be fed from the emergency power system sources (EPS1 and EPS2) and the APU system. The diodes, D1, serve as isolation diodes and prevent back-feeds from the inverters into the emergency power lines. They also prevent unplanned paralleling of equipments. Diodes D2 serve a similar purpose and prevent EPS1 and EPS2 from back-feeding into the 270 Vdc lines from the APU's rectifier assembly. The emergency power lines are additionally protected with current limiters, fuses or circuit breakers, to prevent the emergency power sources from being pulled down by line faults or short circuit faults in any of the individual inverters. To increase further the redundancy of inputs to the DFCS computer, the outputs of the dedicated inverters are cross strapped in pairs as shown in figure 43. Current limiters protect the cross-straps since it is possible for faults to occur in the front end of the DFCS computers themselves.

As configured, the quad redundancy of the DFCS computers can only be compromised by failures within the computers since, even in the event of an all-engine out condition, the four inverters supplying the DFCS computers can still be supplied by the two emergency power supply (EPS) units and/or the APU. In this type of extended emergency the EPS1 and EPS2 will furnish non-interrupt power to the four inverters to maintain flight critical power. On detection of the emergency, the DP/LM system will initiate automatic start-up of the APU and the emergency power sources will, in the interim, power the DFCS system and other flight critical loads, until the APU generator and rectifier come on line. It is possible that there will be more merit in running the APU at light loads, continuously.

Other abnormal and emergency operating conditions are less critical, since power outages of the four inverter power sources cannot be conceived, without the most unusual combination of multiple equipment and power feeder failures. For this reason and the high reliability/redundancy levels, the objective of the 10^{-9} reliability objective for the DFCS and flight critical loads can be achieved.

3.6 Identification of Critical Technologies

It is clear from the trade analysis studies and the description of the systems in the IDEA and the Alternate IDEA that there is significant ongoing work in most of the technology areas that would relieve NASA of the need for further fiscal support. Many of these technologies, such as the VSCF power systems, are already highly mature and they have been the beneficiary of significant military and company funding. The 270 Vdc system is presently without any aircraft implementation, and it is less mature than the VSCF cycloconverter and dc link systems. Similarly, the 20 kHz system is immature, and significant work is required to bring the primary power system concept up to the technology status of the two VSCF systems. There are technology subsets that must be given similar R&D support. The design of the 20 kHz power transmission cables, for example, is one such subset. As stated in the summary/recommendations in the Power Distribution System Trade Analysis, a compendium of cable data for 20 kHz transmission frequencies is required,

along with the development of low loss cable configurations. Studies of EMI, RFI, are also required, as well as further work on the complete integration aspects of a 20 kHz power system in an IDEA and Alternate IDEA based configurations.

In assessing these technologies that require future NASA support and funding, Lockheed has identified those systems and technologies that are presently without substantive background, and/or those systems and technologies that, without NASA support, would not be brought to fruition in a reasonable time frame.

3.6.1 Advanced Power Generation Systems (APGS): - The trade analysis on APGS has already identified the 20 kHz power system as a candidate for continuing NASA support, and this is already being implemented.

The variable-voltage/variable frequency system was also identified in the summary on the APGS as a system that was without major background and one that required significant testing in the laboratories and in aircraft. The VV/VF system will not receive in-house funding from the major electrical suppliers because most of these companies already have significant investment in alternative APGS approaches that they believe to be competitive with the VV/VF. The VV/VF power system is the selected system for aircraft such as the IDEA and Alternate IDEA that have large capacity electric systems, but for implementation it would be necessary for NASA and/or the military to generate design/hardware programs to bring the technology into maturation by the 1990 time frame. There are also subsets of this APGS technology that are discussed separately below, such as the ECS, that will require further and independent development to validate their operation and interface with the VV/VF type power systems.

3.6.2 Emergency Power System Technology. - A more direct subset of the APGS technology is the emergency power system that will be required in the IDEA/Alternate IDEA to support the flight critical (FCS type) loads. Here, the emergency power sources must be examined beyond the level that was possible in the IDEA program to evaluate their efficacy not only from the reliability standpoint, but also from their availability and technology status. It is evident that there are a number of battery configurations, both primary and secondary, that would make highly reliable static power sources to provide non-interrupt power until an APU comes on line, but many of these have other shortcomings: charge difficulties, maintenance problems, logistic support, cost. Also, many batteries have a poor watt-hours/pound characteristic and have low upper-temperature tolerance. Some also, on charge, exhibit thermal runaway problems. Fuel cells and more exotic batteries could be used, but these must also be examined for maintenance/reliability logistic support aspects. On the basis that the emergency will be a rare occurrence, it is possible to consider primary cells with a long shelf life, etc.

An alternative to the static battery power sources are emergency power units, EPUs, of the monopropellant or bi-propellant types. The YC14, F15, F4, Concorde, and other aircraft use liquid propellant EPUs (hydrazine blend) and these have the advantage that these can be long term emergency power sources,

provided adequate fuel capacity is installed. Again, it is another logistic support item (requiring a different type fuel) and it is a somewhat sophisticated, costly installation.

Ram air turbines have the advantage of simplicity. They can also generate power down to reasonably low air speeds and are used in a number of aircraft, including the Lockheed L-1011. These units can be of the drop-out type or can be in a ducted configuration. The RATs have few problems other than that the power-output on low approach speeds can be low, depending on the propeller diameter. For the IDEA, where the flight critical FCS loads require non-interrupt power, the RAT would be non-compliant because it could take 5 to 7 seconds to come on line. Therefore, a backup power source would still be required.

Two alternative versions of the RAT would be to consider power extraction from the bypass fan spool that is normally powered by the engine's low pressure turbine or a vortex-powered generator configuration. In the case of the bypass fan, it would be necessary to evaluate the fan windmilling speed at different altitudes and different air speeds. Provided the emergency generator is permanently tied into (say) the 270 Vdc emergency power supply, this EPS could then provide continuous on-line power and therefore meet the noninterrupt power criterion. These turbines could drive generators that normally support the primary power system and they could also serve as backup to the main engine driven generators, should these fail. Again, the vortex turbine driven generators could provide noninterrupt power, as with the bypass fan driven generator EPS.

Reviewing the above EPS sources, the monopropellant powered systems are now in production and require no NASA support. Conventional batteries such as lead acid, nickel-cadmium, silver-zinc are now in the market-place. The lead acid type is more free of the problems of charging/high cost, etc., but they have low specific output of 8-12 Wh/lb. NASA is therefore urged to pursue the development of more exotic batteries with chemical couples that could provide much higher energy output/lb. Such a program would benefit the space programs as well as the conventional aircraft programs. Continuously fed fuel-cells, such as the solid polymer electrolyte membranes, and advanced versions of the lithium-chloride batteries using hydrogen peroxide as an oxidant could be studied further. Batteries such as lithium sulphur can be projected with maximum theoretical energy densities of 1,300 Wh/lb. Automobile electric-traction systems and air-vehicles would all benefit from a breakthrough in battery technology. Batteries such as the lithium-chloride type with consumable electrodes could become a practical and viable emergency power source for the aircraft application.

Another EPS, not given much attention, but which could be another candidate for NASA consideration, is a stored mechanical energy power system. A high-energy fly-wheel of low volume and high mass could be electrically powered up to speed by an ac motor/generator and, once up to speed with the requisite stored energy the fly wheel could be kept at speed with "topping-power" only from the motor. In the event of an all engine out, and complete loss of primary power, the motor would revert to a "generator-mode" and supply

emergency power for some time, dependent on the kW demand and the energy capacity of the flywheel. A stored energy system of this type could be better than 15 Wh/lb and have better performance than a lead-acid battery; however, if it had to be continuously running for the mission (to provide non-interrupt power) this could be its main disadvantage.

3.6.2.1 Engine starting: This is related to the APGS only because the generators in the IDEA/Alternate IDEA are used in the dual role of a starter. The problems that can be cited against this present proposed method of electric engine starting, derive from two aspects.

(1) The capacity of the power electronics for starting 40,000 pounds SLS rated engines can be in excess of 200 kW.

(2) This power cannot be supplied at the gates of most air terminals around the world. The power is also high in relation to the onboard APUs.

Because of these problems and the fact that the power electronic assemblies have very limited electrothermal capacity, it is recommended that NASA explore alternative methods of electric engine starting that would permit the power electronics to be eliminated from the start cycle, or their power capacity significantly reduced. A corollary to this is that the alternative engine start technology reduce the electric power demand reflected to the APU and external power system. Without such technology, it would be necessary to rely upon APU starts only, since it would not be possible to upgrade the capacity of the electric power supplies at the main airport terminals, in the stipulated 1990 time frames.

3.6.3 Ground Power Support of IDEA. - Because of the large engine starting and ECs power demands, the technology support for an IDEA in the 1990 time frame could be as critical for its impact on the ground power systems (at world air terminals) as the systems are upon the design of the airplane itself. Attention must therefore be given to this critical issue and the appropriate agency (e.g., the Aviation Industry Working Group comprising representatives from AIA, ATA, ICAO, and IATA) should ensure that equally aggressive development programs are put in place to upgrade the capacity of ground electrical power supplies at the air terminals throughout the world.

The airlines presently have tremendous investments in conditioned air and pressurized air ground systems at all main air terminals, to support the pneumatic starting and air cycle air conditioning systems now in the current commercial aircraft. It is axiomatic that without the concomitant development of larger ground electric power systems, the successful integration of the future IDEA type vehicles into viable airline operations could be frustrated by the lack of suitable external power supplies.

Airports will be faced with the problem as to whether to select one or more of the following:

(1) Centralized three phase 200 V 400 Hz installation in each airline terminal. This in turn will break down into two other choices: whether the 400 Hz power be transmitted at 200 V, or at high tension voltage. For voltage drop reasons, the latter is preferable, so step up transformers would be required at the central power station and step down transformers at the satellites.

(2) Distributed three-phase 60 Hz power, where power conditioning (conversion to three-phase 200 V 400 Hz) is accomplished at the satellite: this can be described as a distributed processing (or power conditioning) approach. Again, in this choice, a substation at each airline (or each airport) might distribute three-phase 400 V 60 Hz power, or high tension three-phase 60 Hz power (at say 6,000 V) might be transmitted from the substation to the satellite.

(3) A third issue is whether motor generator sets or static power conditioning is accomplished at the satellites.

(4) Dedicated diesel-driven or other self-contained ground power units located (one or two) per satellite to serve a number of gates (at the satellite).

There is clearly more flexibility with separate dedicated GPUs (ground power units) and it is likely that the costs, compared to major fixed plant facilities, could be much less. In addition, the ROI and the amortization schedules of the portable GPUs could be much shorter (than centralized fixed installations).

3.7 Plans and Resources for Power Generation Technology Readiness

Estimates of the schedule and costs are shown in figure 44 for development of engine-driven generators. Reaching technology readiness will involve the participation of hardware suppliers working with those responsible for integrating systems into the aircraft (usually the airframe manufacturer).

For advanced engine start methodology the schedule is shown in figure 45. This is required for development of new starting methods which will reduce ground power requirements. Technical resources will include APU suppliers and aerospace electrical suppliers. Coordination with engine manufacturers will be necessary for the supply of engine data, together with the role of the airframe manufacturer as systems integrator.

Emergency power (figure 46), requires the resources of electrical suppliers, the chemical industry and battery suppliers, and airframe companies for systems integration.

The implementation of ground power support equipment (figure 47) will require ongoing collaboration between airframe manufacturers, airport operators, and regulatory agencies.

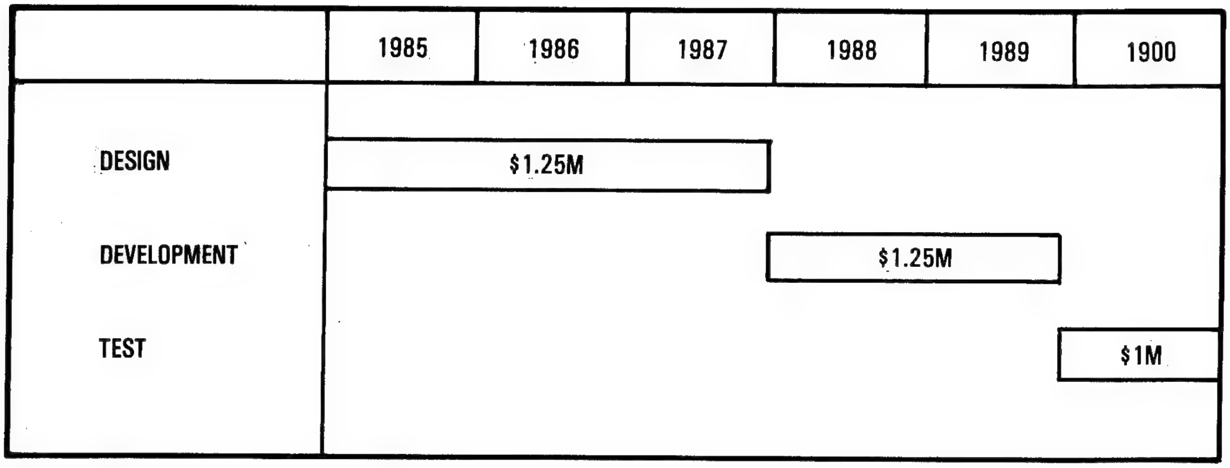


Figure 44. - Advanced power generation.

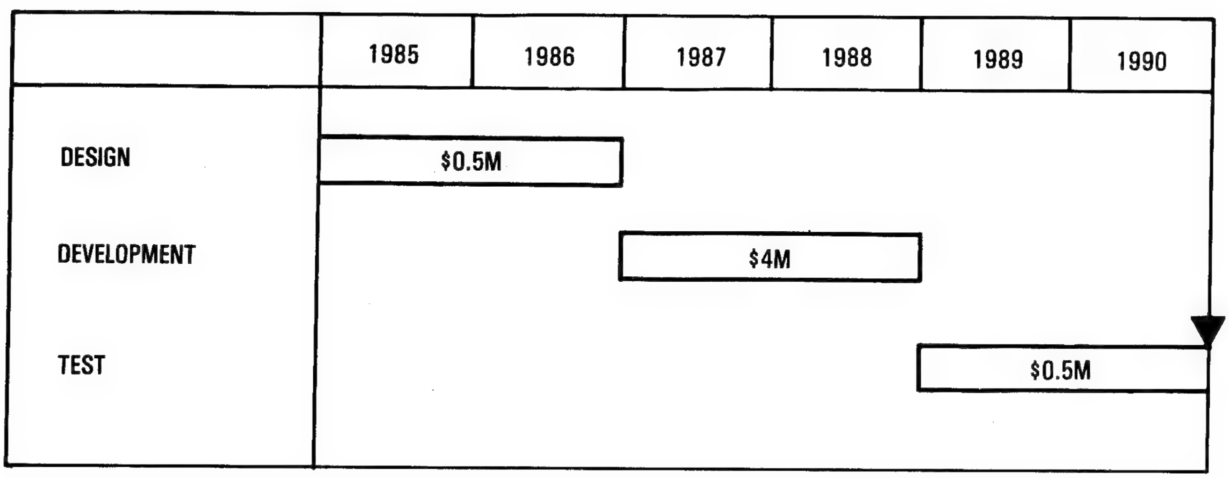


Figure 45. - Advanced engine methodology.

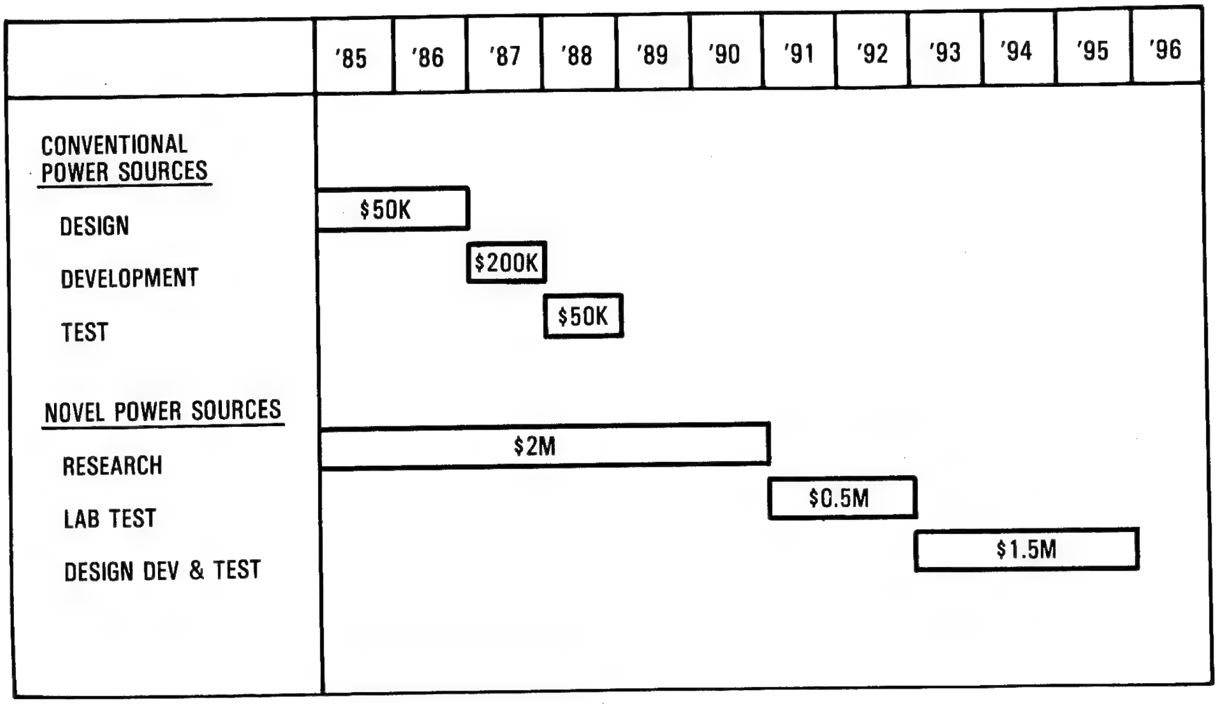


Figure 46. - Emergency power systems, novel power sources.

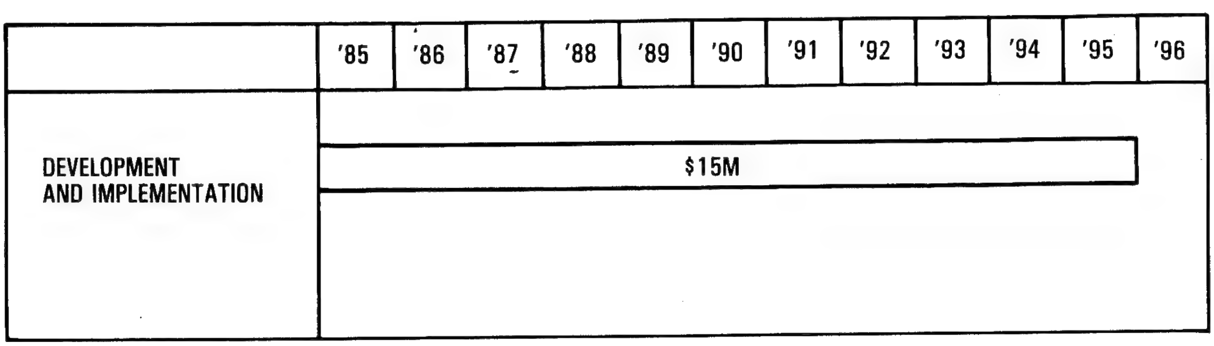


Figure 47. - Ground support equipment.

4. POWER DISTRIBUTION SYSTEM

The power distribution system in a modern aircraft is one of the key systems inasmuch as it determines the layout and routing of the generator power supply feeders, the distribution feeders, and the distribution of the utilization wiring. Typically, the aircraft electrical designer starts with a gross assessment of the electrical load in the aircraft then he assigns various load distribution zones whose locations are determined by any projected major confluences of wiring, e.g., the wing roots, the flight station, the empennage, power plants, or wheel-wells. Power and load management areas are located since these will determine the necessity for and number of main electric load centers (MELCs). These load centers will typically accommodate all the power contactors, voltage-regulators, supervisory panels, engine control/monitoring equipment, insulated/isolated bus-boxes (containing bus-tie contactors, bus contactors, bus transfer inter-ties), and large numbers of rack-mounted load relays and power circuit breakers. Finally, the control center for the total electric system is selected. This usually is the flight station in the large airplane and the cockpit in the military aircraft.

An important design aspect of the power distribution system is the technology and methodology of the protection system for the power generation system, the power supply feeders, the distribution feeders, and the utilization wiring. In most modern aircraft load protection is furnished by electrothermal circuit breakers of the trip-free type, which are circuit-breakers that cannot be overridden when they are in a tripped state. In some aircraft, "current limiters" were used which were high rupturing capacity fuses of the "slow-blow" type that protected the distribution feeders and allowed protection coordination with load circuit breakers located downstream at various remote load centers (RLCs). The primary shortcoming of this type of protection methodology was that the remotely located circuit breakers which provided the necessary protection of the load wiring against smoke/fire hazards required manual reset, so physical access to the circuit breakers was necessary. This access was often difficult and, in some cases, impossible. As a result, such remotely located circuit breakers could only be used in the low priority load circuits. Implicity, all other circuit breakers for the control and power wiring of all loads, other than the low priority circuits had to be in the flight station. As a consequence the flight station in aircraft such as the L-1011, C-141, C-5, DC-10, and 747, became cluttered with more than 1000 control and power circuit breakers.

Power supply wiring, the large gage feeders between the power plant located generators and the MELCS, requires special consideration and protection treatment. This usually involves the employment of differential zone-type protection so current transformers are located at the "sending end" of the power feeders near the generator terminals and an equivalent set are located at the "receiving end" of the power cables where they enter the MELCs. Since these CTs monitor the current at both ends, it is possible to compare the currents at the "sending" and "receiving" ends of the cables and initiate a generator-trip when any difference current is detected. For aircraft safety reasons, the de-energizing of the zone-protected feeders is accomplished in the shortest possible time and typically in less than 25 millisecs. The motivation for the trip times derives from the concern that cable shorts can

create sparking conditions in the power plant plants and/or wings, where the possibility of flammable vapors exists. In addition to the zone-protection CTs, other CTs are used for flight station/ cockpit instrumentation/indicators and for the generator protection/supervisory panel, etc.

4.1 Power Distribution Systems Alternatives

The power distribution systems alternatives considered for the IDEA configuration are:

- Radial Distribution
- Distributed Bus
  - Common Duplicated Ring Bus
  - Separate Duplicated Ring Bus
  - Duplicated Selected Radial
  - Hybrid Ring/Distributed Bus

4.1.1 Radial Distribution. - One of the requirements in this study is to evaluate alternative power distribution systems for the application in the IDEA and Alternate IDEA. The Baseline itself is considered to be a conventional radial distribution of the type described above. These types of systems can be schematically illustrated by figures 48 and 49, which simply define the routing of the power supply and distribution feeders from the generators to the RLCs in a two-engined airplane. In figure 48, a main electric load center (MELC) is assumed to be located in the flight station/ cockpit area and the power feeders from the Nos. 1 and 2 engine driven generators are taken directly to the flight station (FS). Differential-protection is used for these power feeders.

At the flight station, two isolated buses are established. In an emergency, they can be connected by the bus tie contactor, BT1: flight station loads, FSL, are connected to the buses. From the MELC distribution feeders are routed to RLCs at the:

- Left wing root area
- Right wing root area
- Empennage

Protection of the distribution feeders is by high rupturing capacity fuses but electrothermal units could also be used. Single-wire distribution is used, but in the case of a three-phase ac system there might be one or more wires per phase. At each of the RLCs, a large number of load wires will supply the left wing loads (LWL); the right wing loads (RWL); the FS loads (FSL); and the tail loads (TL). Protection for this load wiring will usually

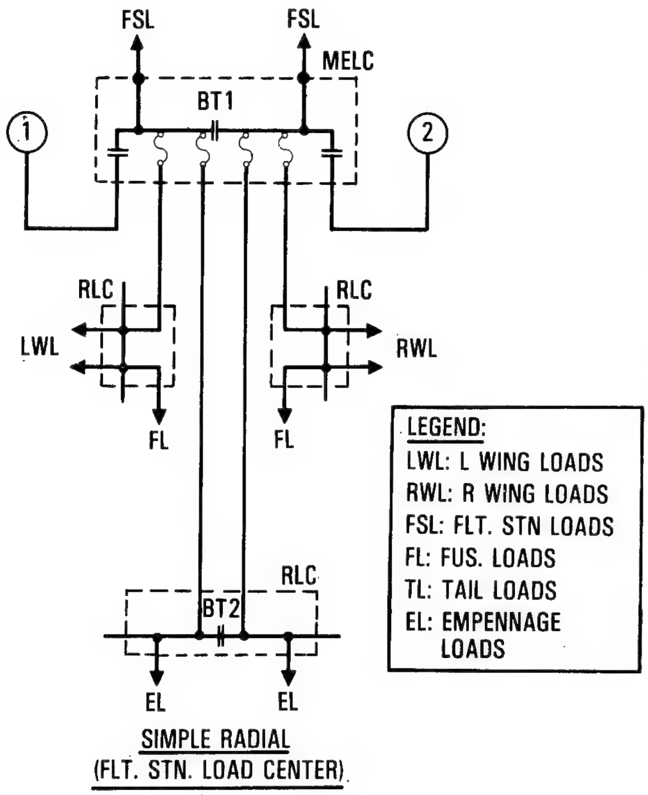


Figure 48. - Power distribution systems: radial (flight station load center).

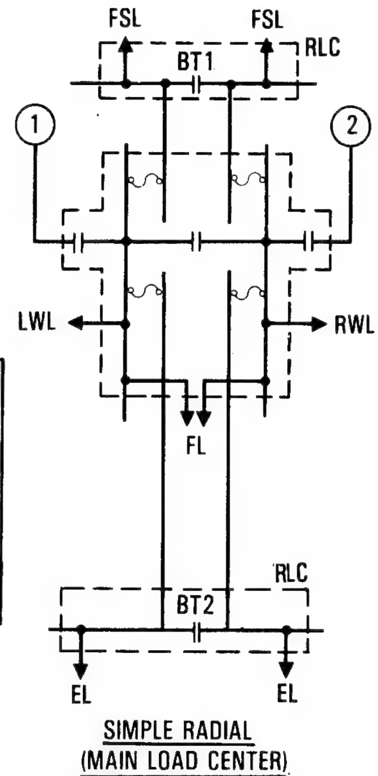


Figure 49. - Power distribution systems: radial (main load center).

be electrothermal or electromagnetic circuit breakers since access to such circuit breakers is not necessary in flight. While not shown in the schematic, significant amounts of electric/electronic equipment, and relays/contactors, will also be located in the MELC. Where the power supply is alternating current, step down transformers power the low voltage loads, such as lighting as well as the transformer-rectifier units that furnish 28 Vdc power.

Figure 49 is a simple variant of figure 48 and it is still a two-engine radial distribution system with the following basic differences:

- The MELC is centrally located near the wing root.
- Protected feeder system is used in the RLCs.

Shown as a dc rather than an ac system, three-phase wires are used to supply power to each RLC and each of these wires is protected at the sending end by current limiters or circuit breakers. As before, load wires distribute from the MELC and the RLC to the wing and fuselage loads. These load wires are protected by the typical trip free circuit breakers.

4.1.2 Ring and Distributed Bus Systems. - By definition, a distributed-bus system is differentiated from a radial-system because subbuses and/or individual loads may be tapped into the distributed bus as it traverses the aircraft. As indicated in the overview to this discussion on power distribution/protection methodology, radial power systems have the problem that a very large number of loads are powered over many wires emanating radially from the MELC and/or the RLCs. The proliferation of wiring and the unusually large number of circuit breakers are, therefore, cited as the disadvantages of the radial type distribution system.

Two aspects attract consideration of the distributed bus concept:

(1) The large increase in the number of loads in the wings and wheel-wells in the IDEA and Alternate IDEA airplanes.

(2) Solid state power controllers (SSPCs) and remote power controllers and (RPCs) make possible locating the controllers in close proximity to the loads. This eliminates the need for locating all the contactors/relays/SSPCs in a centralized fuselage location. It also eliminates the weight penalty of using a large number of load-dedicated wires that are much longer in length than wires tapped from the distributed bus to the loads.

Implicit in the above is the fact that each SSPC, RPC be they fully solid-state or hybrid, combines the features of a circuit breaker and contactor relay. More importantly, it is implicit that such control/protection devices are of the automatic trip type that can be reset from the flight station or automatically by the load management system (with appropriate monitoring by the crew).

Ring and combination ring/distributed bus systems constitute a basic departure from the radial and the radial/distributed bus system in that power from the generators ties directly into the ring distribution system so the feeders are not taken into one or more MELCs in the fuselage. This means that the zone of differential-protection is shortened for the generator/power supply feeder system, but such protection will be retained for these alternative approaches to power distribution. However, in the IDEA and all-electric aircraft in general, a different focus may be given to the protection methodology for the generators and power feeders in these aircraft. For example, the most prevalent faults in conventional aircraft power systems are line-to-ground faults, which give rise to very high and sometimes destructive fault currents. These fault currents may be typically limited by the impedance of the fault, the impedance of the line and the source impedance of the generator. A full analysis of symmetrical and asymmetrical fault currents in an electric system is beyond the scope of this study as it involves a definition of the positive, negative and zero sequence values of the cables and generators. Also, the amplitude of these fault currents are affected by the per phase cable groupings, their number/configuration and their stand-off clearance from aircraft structure. Of interest here is the fact that these standoff distance parameters are more pertinent to aircraft using conventional aluminum, or advanced metallic structures, such as the lithium alloys, rather than composite structures. Clearly, the latter will involve a different computer modeling of the generators with their feeder distribution systems.

In addition to the line-to-ground faults, there is the problem of incipient line-to-line fault and, while these have a lower statistical prospect than line-to-ground faults, they must nonetheless be considered. Also, unlike the line-to-ground faults, the line-to-line faults are more likely to occur at equipment terminations rather than in a run of cable. Therefore, very careful attention must be paid to the method of cable terminations and their interface with contactors, relays, SSPCs/RPCs and other equipment/component assemblies. Unfortunately, in many aircraft electrical installations phase-barriers between cable lugs/terminations have been inadequate or of poor design and insufficient-torquing of the studs or screws has resulted in vibration causing the terminations to loosen. The result of this is that the termination contact-resistance has increased to a point that the ensuing heat generation caused carbon-tracking across the plastics and, later a problem of line-to-line shorts led to severe arcing and fire hazard. The prospects of line-to-line fault must, therefore, be addressed at the equipment terminations with due recognition of the problems involved. Also they must be given special attention in the IDEA, since this aircraft will clearly be more dependent upon the reliability and integrity of the generator/feeder wire distribution system.

The paralleled system exacerbates the rupturing current problems for the switch-gear and other protective devices as do large, stiff aircraft power systems; as a consequence, the fault currents might be in the range of 5000 to 15,000 amperes. For the nonparalleled system, the line-to-ground fault currents might be approximately a maximum of 700 to 1250 amps. For the IDEA, the utilization of higher voltages, such as a three-phase 400 V system, would reduce these currents by approximately 50 percent.

In aircraft power systems fault currents are also a direct function of the ceiling voltage on the generator and its electrical stiffness relative to short circuit conditions. Therefore, with permanent magnet generators the transient maximum fault current amplitude will (in the VSCF type power system) be controlled by the current limiting control technology in the converters/inverters. In the 270 Vdc systems the phase delay rectifiers will control the fault currents typically the phase angle on the SCRs will fully advance on short-circuit and create a short-time overvoltage/overcurrent condition. In the variable voltage/variable frequency system the voltage will not increase, and will actually decrease as a function of the cross (demagnetizing) ampere-turns and the machine/cable reactances. For wound rotor machines the peak fault-currents will be a function of the maximum excitation that can be impressed upon the generator rotating field system.

It is clear that several factors affect the amplitude of the fault currents in aircraft power systems and these, in turn, affect the sizing of the power supply/distribution feeders. The objectives are to reduce the prospective fault currents and to ameliorate the switchgear problems. Beyond this, it is possible to consider ungrounding generator neutrals and allowing them to float. Alternatively, the neutrals could be grounded through high impedance relays that could serve the dual purpose of limiting the line-to-ground faults to a matter of milliamps and providing a ground fault warning signal to the power management system at the same time.

4.1.2.1 Common duplicated ring bus: This system is shown schematically in figure 50. It consists of two complete common distributed ring buses, which are tied or isolated at their extremities by section contactors. All contactors are normally closed except flight station and tail contactors and are under control of the microprocessor which is a part of the power management system. For example, on the ground both of the duplicated ring buses can be formed into two complete rings in which either or both ring buses can be powered up from the APU or the external power supply. Again, the nomenclature is as before.

FSL:	Flight Station Loads
LWL:	Left Wing Loads
RWL:	Right Wing Loads
EL:	Empennage Loads
LSB1/LSB2:	Left Section Breaks 1 and 2
RSB1/RSB2:	Right Section Breaks 1 and 2
TSB1/TSB2:	Tail Section Breaks 1 and 2
FSSB1/FSSB2:	Flight Station Section Breaks 1 and 2

During normal flight operation the wing section breaks are closed and the flight station/tail section breaks are open. Therefore, as shown, the No. 1 generator supplies the left wing and left fuselage buses while the No. 2 generator supplies right wing and right fuselage buses.

In the event of a No. 1 engine or No. 1 generator failure the section breaks remain as before, except that the flight station section breaks (FSSB1 and FSSB2) and the tail section breaks (TSB1 and TSB2) close. At this time both distributed ring buses are complete rings and are supplied by the No. 2 generator. Clearly, this latter situation is the same in the event of a failure of the No. 2 engine or generator except that both ring buses are then supplied by the No. 1 generator. All loads may be conveniently tapped into the distributed ring buses as they traverse the periphery of the aircraft. Unlike the radial system, the load power wires are short and direct; however, it is important that the integrity and reliability of the distributed ring buses be of the highest order. Therefore, since any short circuits in the load wires pose a threat to the buses, it is essential that protection for each load wire be effected as close to the bus as possible. Also, as highlighted earlier, it is necessary that the number of breaks and terminations be a minimum.

4.1.2.2 Separate duplicated ring bus: This distributed ring bus system is of the same genealogy as the common duplicated ring bus system, except it falls more precisely into the definition of a ring bus system. This is shown more clearly by figure 51 where it can be seen that two closed rings are provided on each wing and two closed distributed rings are used in the fuselage. The system is also simpler and uses fewer section break contactors.

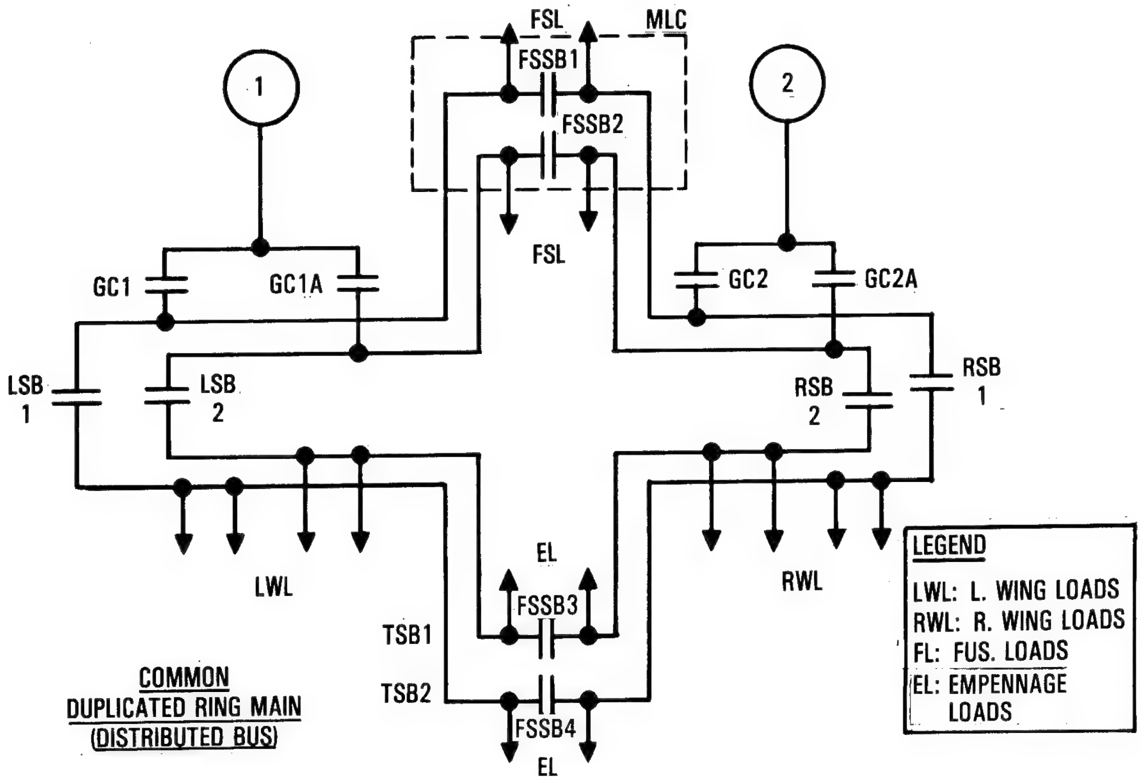


Figure 50. - Power distribution systems: dual ring mains (common).

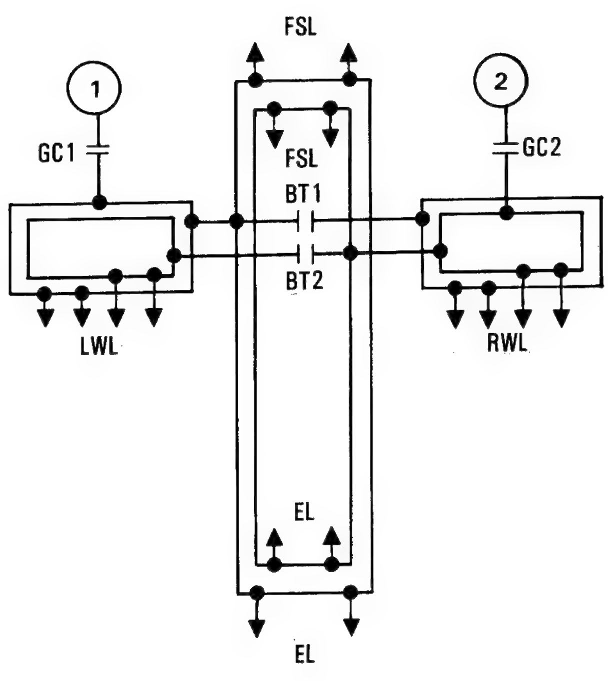


Figure 51. - Power distribution systems: dual ring mains (separate).

Operationally, the separate duplicated ring system is not much different from the common duplicated ring bus system. The No. 1 generator supplies the left outer wing bus, the fuselage outer bus, and right wing outer bus via BT1. Conversely, the No. 2 generator supplies the right wing inner ring bus, the left wing inner ring bus via BT2, and the fuselage inner ring bus.

In the event of a failure of generator 1 or 2 bus ties BT1 and BT2 close to permit the remaining generator to supply both fuselage buses and both wing buses.

4.1.2.3 Duplicated selected radial: This is a straightforward radial power distribution system that offers the advantage of a distributed bus system and provides dual-redundant power feeders in the wings and fuselage. The schematic configuration is shown in figure 52, where the dual redundant feeders are provided at the source-end in a split MELC having two insulated/isolated source buses. Each of the source buses is selectively switchable to the other bus, under the power management processor control, and all feeders emanating from the buses are protected with fuses or circuit breakers. The aspect of selectability is carried through into the wings and fuselage where local buses (or individual loads) can be connected to either of the radial distributed buses.

In this type of system, certain low priority loads are permanently connected to one or other distributed radial bus and provide switchability only for the high priority loads, either individually or by a sub (essential) bus. The only time the high priority loads would have additional power source access is where there is a failure of one or the other distributed bus to ground. As can be noted from figure 52, if there is only a failure of an engine or a generator, the source bus in the MELC is transferred via a double-throw power contactor to the other source bus. Notwithstanding this advantage that accrues also to the low priority loads, a load hierarchial structure is set up for all the loads in the aircraft. All loads are assigned different levels of prioritization and will be disconnected from the electric power system in case of an emergency, in accord with their priority status.

The duplicated selected radial is a simpler system from the standpoint of routing the redundant power feeders in the aircraft, but it engenders more complexity than the "connectable" ring systems since there is a proliferation of load and/or bus switching in the wings and fuselage.

4.1.2.4 Hybrid ring/distributed bus: This system is shown schematically in figure 53. As in all hybrid systems, the intent is to combine the features and advantages of the two types of power distribution concepts.

As shown, duplicated ring buses are provided in each wing and four isolated, redundant, distributed buses are provided in the fuselage. Another notable difference is that a three-engine airplane configuration and the interface with the APU and external power is shown.

The design layout of the hybrid ring/distributed bus system follows the practice of the previously described two-engine systems. However, as there are two generators per power plant, it is possible for the each ring in each

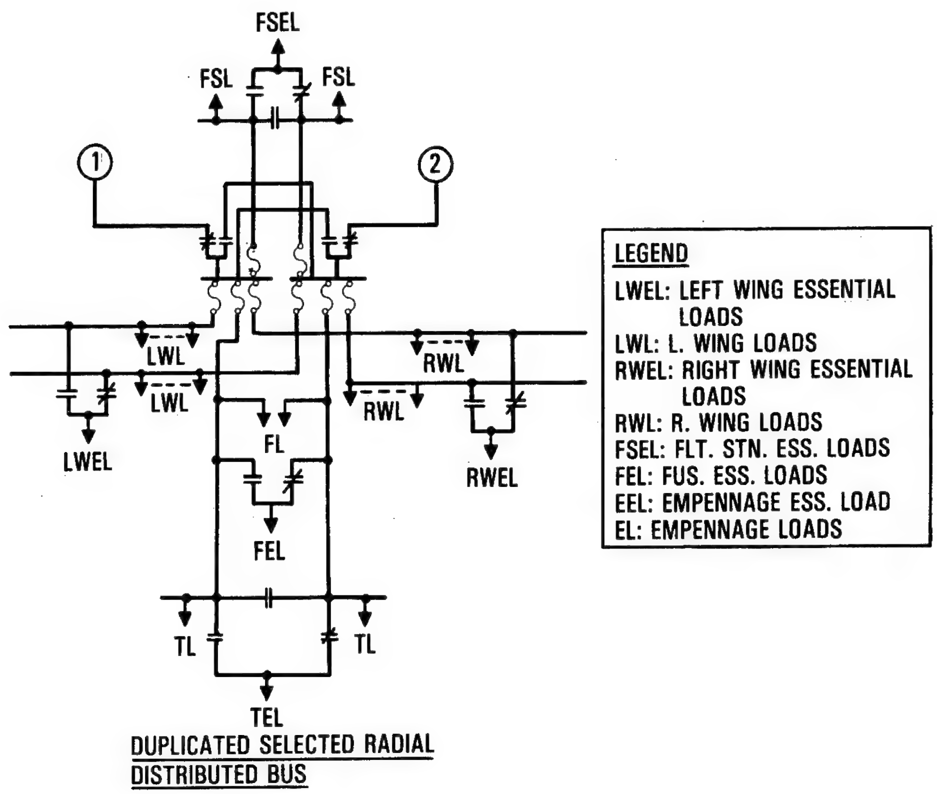


Figure 52. - Power distribution systems: duplicated selected radial.

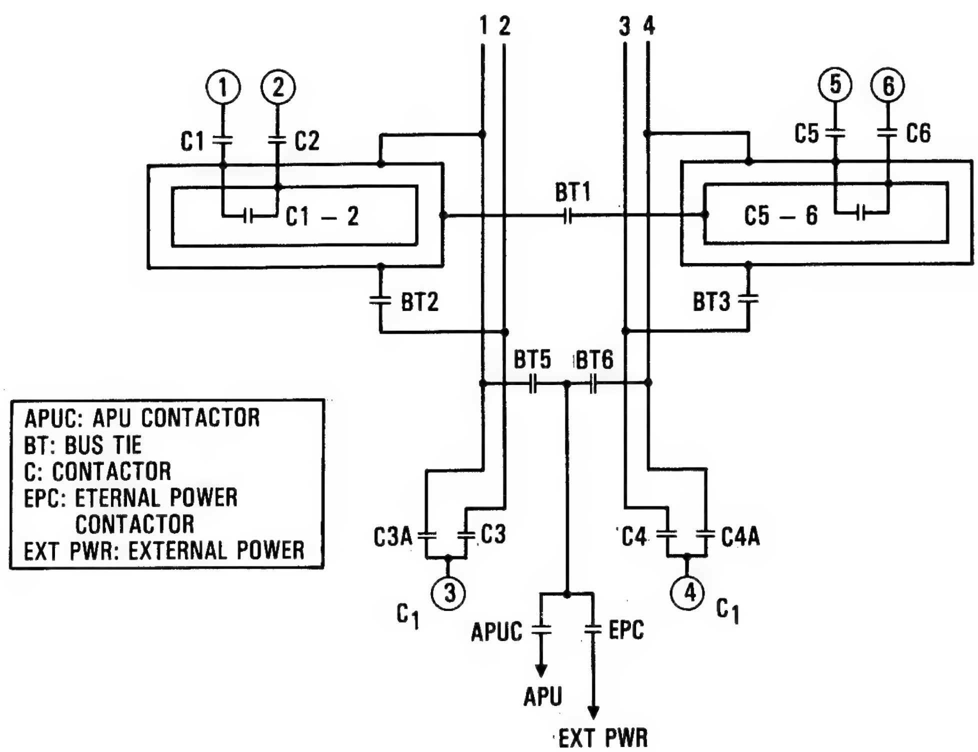


Figure 53. - Power distribution system: ring main (three engine airplane).

wing to have the luxury of being powered by its own generator or either generator can power both buses in the event of failure of the other, e.g., contactor C1-2 can tie the duplicated rings in the left wing. Another feature is that the outer rings of each wing are also powering the outer pair of distributed feeders (1 and 2) in the fuselage, while the other two fuselage distributed feeders 3 and 4 are powered from generators 3 and 4 via contactors C3 and C4, respectively. Further, it can be seen that the outer ring (powered by say No. 1 generator) is a backup (via BT2) for the No. 2 fuselage distributed bus in the event that the No. 3 generator fails. Note that No. 3 generator normally powers the No. 2 distributed bus via contactor C3. This modus operandi is symmetrical on the right side with the left side inasmuch as the configuration of the right wing duplicated rings and the empennage-located No. 4 generator mirror the left side.

Figure 53 system again enjoys a basic simplicity in that it is relatively uncomplicated system, despite the fact that it is a 3 engine/6 generator configuration. Also the system provides separate, isolated, quad redundant feeders in the fuselage. This quad redundancy is considered essential in the IDEA because quad redundant electric actuators and quad redundant FCS digital computers are required for the flying tail (the horizontal stabilizer).

Interface of the APU/external power systems is also simple in that the fuselage feeder configuration is selectable to the APU or external power and it feeds into the center top point of bus ties, BT5 and BT6. With this simple arrangement it is possible for one single power source to power every bus (ring and distributed) in the airplane. Implicitly, this same operation is possible in flight, when the APU or any one of the six generators can also power all or any combination buses. The control of the section breaks and contactors will, of course, come under the purview of the power management system.

4.1.3 Weight assessment. - This weight assessment is made on the 3-engine/6-generator type systems that will be used in the IDEA and Alternate IDEA airplanes. A comparison of the weight of the systems in these two all electric aircraft with the Baseline system is, as stated, not meaningful since the power distribution capacities are completely different. However, the weights of the proposed power distribution are discussed below, based on the predicted capacities of the generation systems in each aircraft.

Baseline	3 x 90/120 kVA
IDEA	6 x 150/200 kVA
Alternate IDEA	6 x 220/275 kVA

The hybrid system of the figure 53 configuration is used for the weight assessment, with distributed ring buses in each wing and quad redundant distributed feeders in the fuselage. Figure 54 is illustrative of the lengths of the power feeders in each wing and the fuselage.

Weights of the distribution feeders are premised on maintaining the same size conductor throughout the airplane, even though it would be possible at certain routing points to downsize the cross section of the power feeders. This, however, would require more breaks and terminations at those points

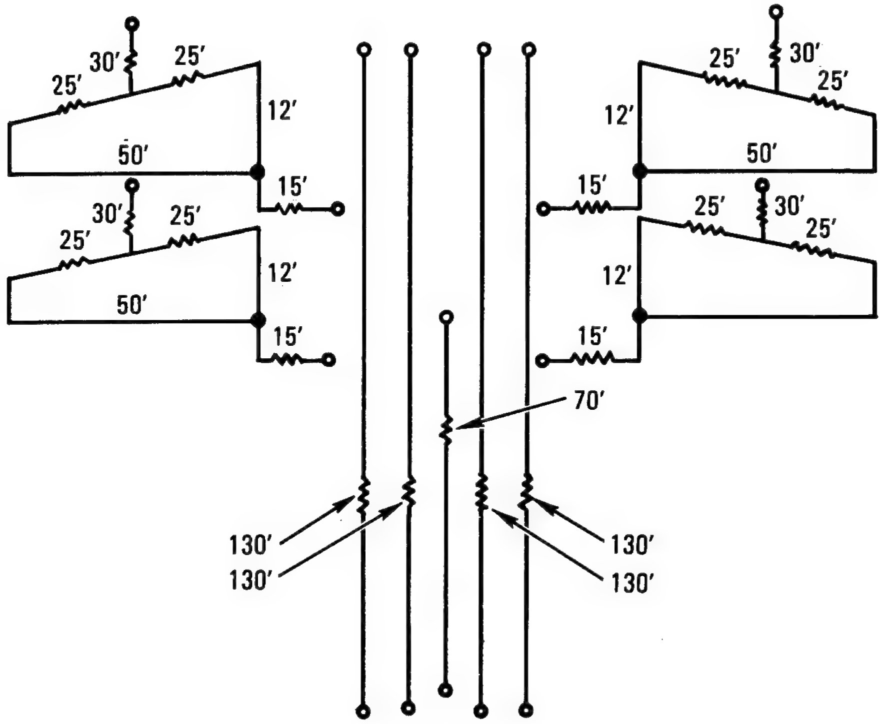


Figure 54. - Feeder length/impedance diagram.

where the wire gage changes were made. The weight advantage of doing this is considered marginal and less attractive relative to the objective of achieving high reliability in the transmission system. The other sizing criterion is that every feeder be sized for the base-rating of the generator and not the higher slash-rating. The rationale for this is that only one generator can supply a feeder section at a time, and in the event of overloads, advantage can be taken of the inherent overload capacity of the cables' high temperature insulation system. Thus, it is projected, with the infrequent prospect of this occurrence, that the impact on the insulation life of the cable insulation system will be insignificant.

With the above premises the cable rating of the three-phase 400 V 150 kVA system in the IDEA airplane will be:

$$I_L = \frac{\text{kVA} \times 10^3}{3 \times V} = \frac{150 \times 10^3}{3 \times 400} = \underline{216\text{A}}$$

Cable Data:	Select	0 gage
	Weight	378 lb/1000 ft
	Resis.	0.114 ohms /1000 ft
	Wire Dia	0.525 in

Cable Lengths: See Figure 47

Wings:

Cable Length/Wing: = [30+2(50)+12+15]x2=314

Cable/2 Wings: = 2x314x3*=1884 ft

Fuselage:

Cable/Fuselage Side = 20+(2x130)=280 ft

Cable/Fuselage (2 sides) = 2x260x3*=1560 ft

Empennage:

APU Feeders = [70+2(12)]x3*=282 ft

Aircraft Power/Distribution Feeders

Total Length: L_c = 1884'+1560'+282'=3726 ft

Total Cable Weight:

$W_c = 3726 \times 378 \times 10^{-3} = 1408$ lb

* Three-phase cables.

4.1.4 Feeder voltage drop. - It is appropriate to comment on whether the wire gages used in the foregoing calculations are volts drop limited or current limited. It is always possible to find a wire gage that is adequate for the current, but the wire gage in long-line transmission systems may have to increase over the gage necessary to carry the current because of voltage drop considerations. Also the dc resistance of the wire has been used rather than its ac impedance, which is affected by the cable resistance and reactance at the transmission frequency of interest. The cable therefore has a power factor and it tends to be lower as the frequency increases.

Using the dc resistance values in the previous calculations, the following voltage drops would be appropriate the 100 feet of wire carrying the wires rated current.

	kVA	Rtd Amps	Wire Gage	Wire Resis/1,000 ft	Wire Voltage Drop/100 ft
Baseline:	90	250	00	0.09	2.25
IDEA:	150	208	0	0.114	2.37
Alternate IDEA:	220	305	2x1	$\frac{0.146}{2}$	2.22

Considering only the IR drop value, it is clear that none of the above cables is voltage drop limited since they are well within the permitted 5 percent volt drop limit of typical three-phase ac systems. For example, the

conventional three-phase 200 V 400 Hz generator transmits 208 V at the terminals allowing an approximate 4 percent line voltage drop. Also for a nominal 400 V system the generator would have a terminal voltage of 416 V approximately. A significant advantage of the higher voltage system is that it allows the cable impedance to increase as a square of the voltage ratio for the same percent voltage drop. Figure 55 is illustrative of the impact of voltage drop on cable sizing. It shows the sensitivity of the lower gage wires to cable rating vs distance for a permissible 4 V drop. Note that the current rated value of 9 A for a 22 gage cable can only be carried for a distance of 27 feet. Therefore, if a run of 100 feet were involved, it would be necessary for a higher gage cable to be selected. Cable size is determined by selecting the first cable gage line below the intersection of the projections of the desired current capacity and distance values. Reference to Figure 55 shows that, for the present example, a 16 gage cable would be required. This emphasizes the penalty in using radial type power systems.

If cable impedance rather than just IR drop is taken into account, the appropriate vectorial impedance voltage drop must be considered in accordance with figure 56. This vector diagram, is based on the following formula:

$$\text{Voltage Drop, } BE = IR\cos\theta + IX\sin\theta + (IX\cos\theta - IR\sin\theta)^2/2V$$

Figure 56a illustrates the typical cable drop as derived from the above formula. Simply, the sending end (line) voltage is OA and the receiving end voltage is OB. The voltage drop that is calculated is $OE - OB = BE \simeq BD$: BD now consists of in-phase components BC and CD (where $BC = BF\cos\theta$ and $CD = AF\sin\theta$: the small component DE is given by the $(IX\cos\theta - IR\sin\theta)^2/2V$ part of the expression. The values of figures 56b and 56c are that they illustrate the impact of the load power factor upon the cable voltage drop. Figure 56b shows that if the line current I_L lags by a large angle (low power factor), the IX component is significant in reducing the receiving end voltage V_2; however in figure 56c, where the phase displacement is smaller, the IX component has marginal effect, but the IR component is (relatively) more effective in reducing V_2. Therefore, based on the generalization inherent in figures 56a and 56b, it can be said that considering the large gage cables in the IDEA and Alternate IDEA, the cables are unlikely to be volts-drop limited.

It is recognized that it is the cable inductance, as a function of frequency, that is pertinent. In this regard, the following formula relates the cable inductance versus other variables.

$$\text{Inductance, } L = 0.1401 \log_{10}\left(\frac{2301\ D}{\sqrt{CM}}\right) \text{(in millihenries/1000 ft)} \qquad (4\text{-}1)$$

where

2301 is a factor for a 19 strand conductor

D is spacing between conductors (in inches)

CM is circular mil area of conductor

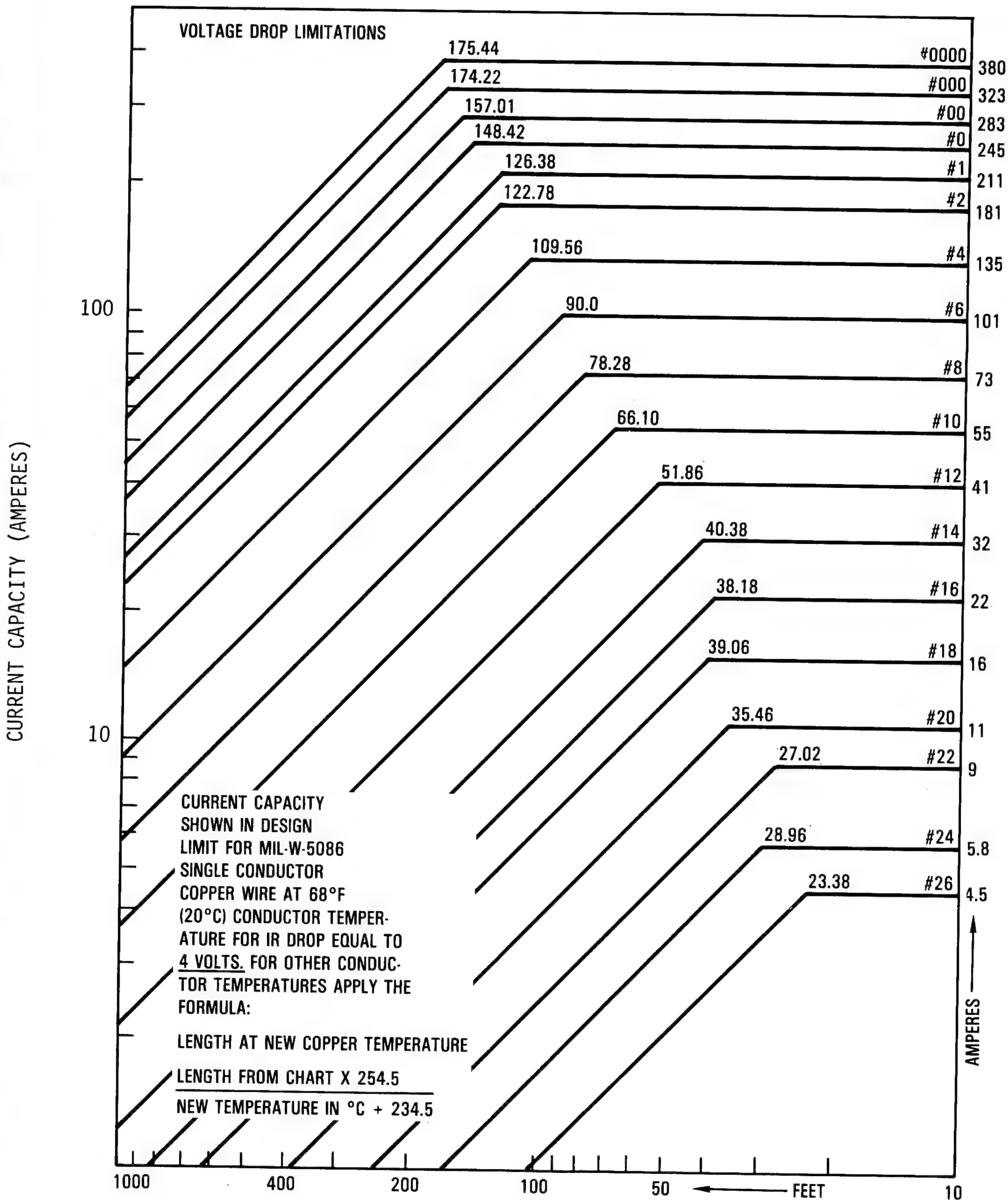


Figure 55. - Cable size/voltage drop vs distance.

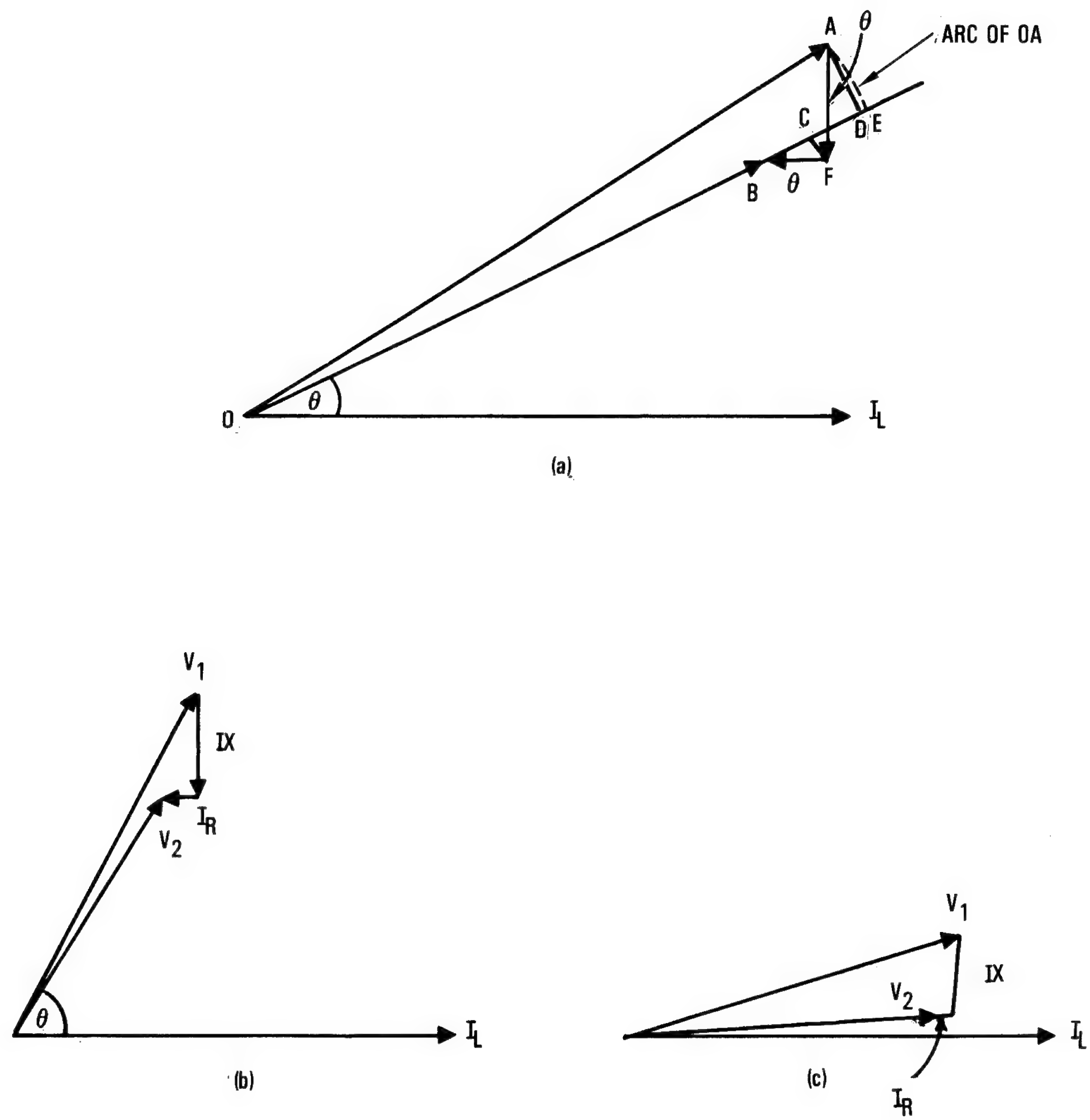


Figure 56. - Vector voltage drops for ac transmission cables.

Since the cable reactance, $X_L = 2\pi fL$, the expression becomes:

$$X_L = 2\pi\ f \times 0.1401 \log_{10}\left(\frac{2301\ D}{\sqrt{CM}}\right) \text{ ohms/1000 ft}$$

or, for 400 Hz

$$X_L = 0.352 \log_{10}\left(\frac{2301\ D}{\sqrt{CM}}\right) \text{ ohms/1000 ft}$$

In most tables showing cable data it is possible to select cable sizes based on their current carrying and to determine whether such cables are assessed as having a "free air" or "bundle" rating. The tables also show the physical data on cables such as their specific weight (lb/1000 ft), conductor-diameter and finished-wire diameter. More typically, however, the cable resistance/1000 feet as shown is the cable's dc resistance, and this is inadequate in ac systems where it is necessary to know the impedance value (the R + jX value) for a given frequency, wire grouping configuration and the standoff distance from structure. Table 9 is typical of a tabulation of such data for 400 Hz system, while figure 57 shows the cable impedance vs transmission frequencies up to 4 kHz. Figure 58 further shows the cable power factor changes as a function of different gage cables at 400 Hz and 4 kHz.

It is through the use of such data that more detailed analyses can be made of any new cable transmission system in an advanced aircraft.

A corollary to the consideration of the R + jX effect in ac transmission cables is the impact of frequency upon "skin-effect." Skin effect is a phenomenon in ac systems, where the current tends to confine itself to the outer periphery of the conductor. It is an effect which increases as the frequency increases. The formula that relates the significant variable is given below.

Skin Effect $\quad X = 2\ a\sqrt{\frac{2\ \pi\ f}{\rho}}$

where

a = conductor radius (cm)
f = frequency (Hz)
μ = permeability
ρ = resistivity (ab ohm-cm)
X = factor defined in figure 59

The physical and practical aspect of skin effect is that it increases the "R" value of the R + jX impedance because the effective cross-sectional area is parasitically decreased. Therefore, it can be seen that high frequencies will exacerbate the skin-effect and increase the losses in the transmission cable, unless special precautions are taken. However, more practically, and for the more typical aircraft frequencies, it is possible to decrease the skin effect by using smaller gage wires, since the large gage conductors tend to exhibit higher reactance in the inner core and so the current tends to crowd

TABLE 9. - POSITIVE SEQUENCE IMPEDANCE FOR EQUILATERAL CONFIGURATION

Wire Size (MIL-W-5086)	Impedance in Ohms per 1000 ft.	
	Min. Elevation*	Six Inch Elevation*
AN-0	0.106+j0.182	0.101+j0.185
AN-2	0.165+j0.192	0.158+j0.195
AN-4	0.262+j0.201	0.256+j0.201
AN-6	0.413+j0.218	0.409+j0.220
AN-10	1.018+j0.227	1.014+j0.227
AN-14	2.507+j0.260	2.503+j0.263
AN-18	5.531+j0.286	5.529+j0.286
AL-0	0.165+j0.192	0.158+j0.196
AL-2	0.255+j0.198	0.253+j0.202

*Elevation of three conductors above aircraft skin

its outer elements of the conductor. The cable ac resistance is related by the formula

$$R_{AC} = k\ R_{DC} \tag{4-4}$$

Where

R_{AC} is the ac resistance of the cable

R_{DC} is the cable's dc resistance, (obtained from Tables)

k is the skin effect ratio as related to X by curves such as figure 59

Another aspect of the ac cable evaluation is the cable power-factor effect. Here again the larger cables are adversely affected as a function of frequency inasmuch as the cable power factor is lower for the larger size conductors. This effect is shown in figure 58, where for example at 400 Hz, a #10 gage wire has a power factor compared to an #0 gage cable with a power factor of about 0.42.

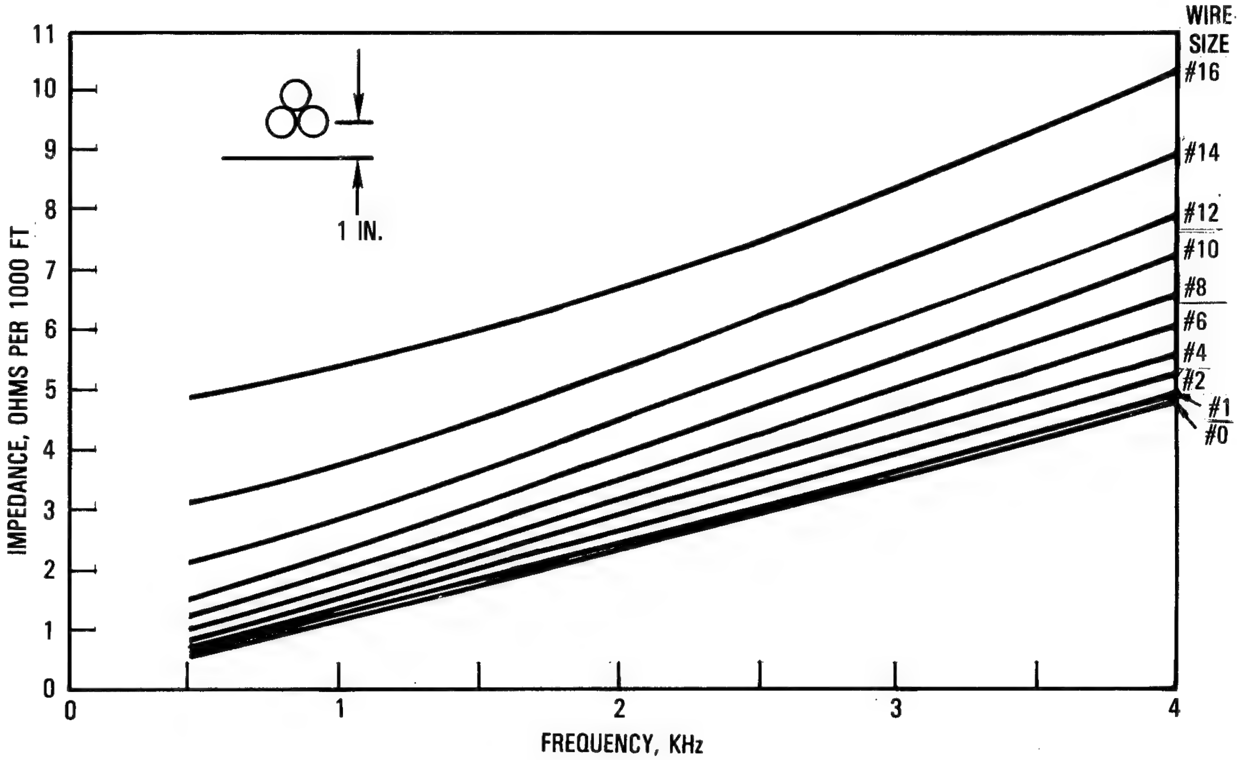


Figure 57. - Cable impedance vs frequency. (Courtesy AiResearch)

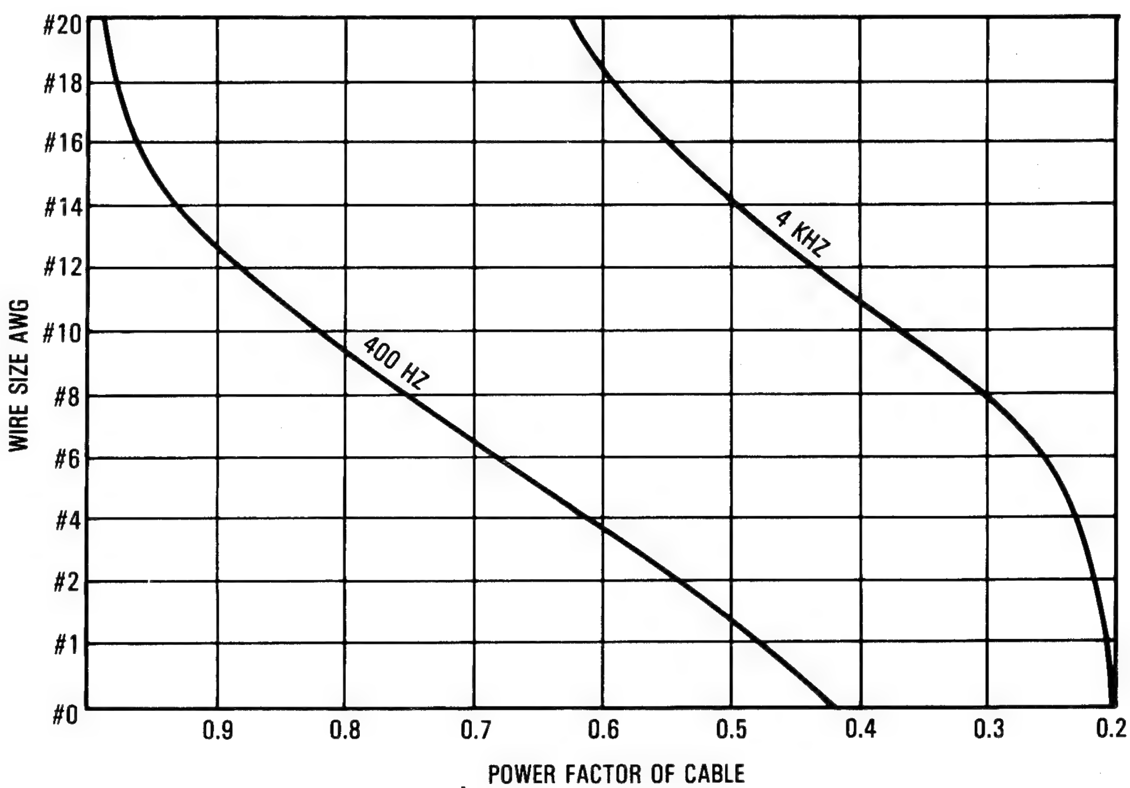


Figure 58. - Cable impedance power factor at 400 Hz and 4 kHz for various cable sizes. (Courtesy AiResearch).

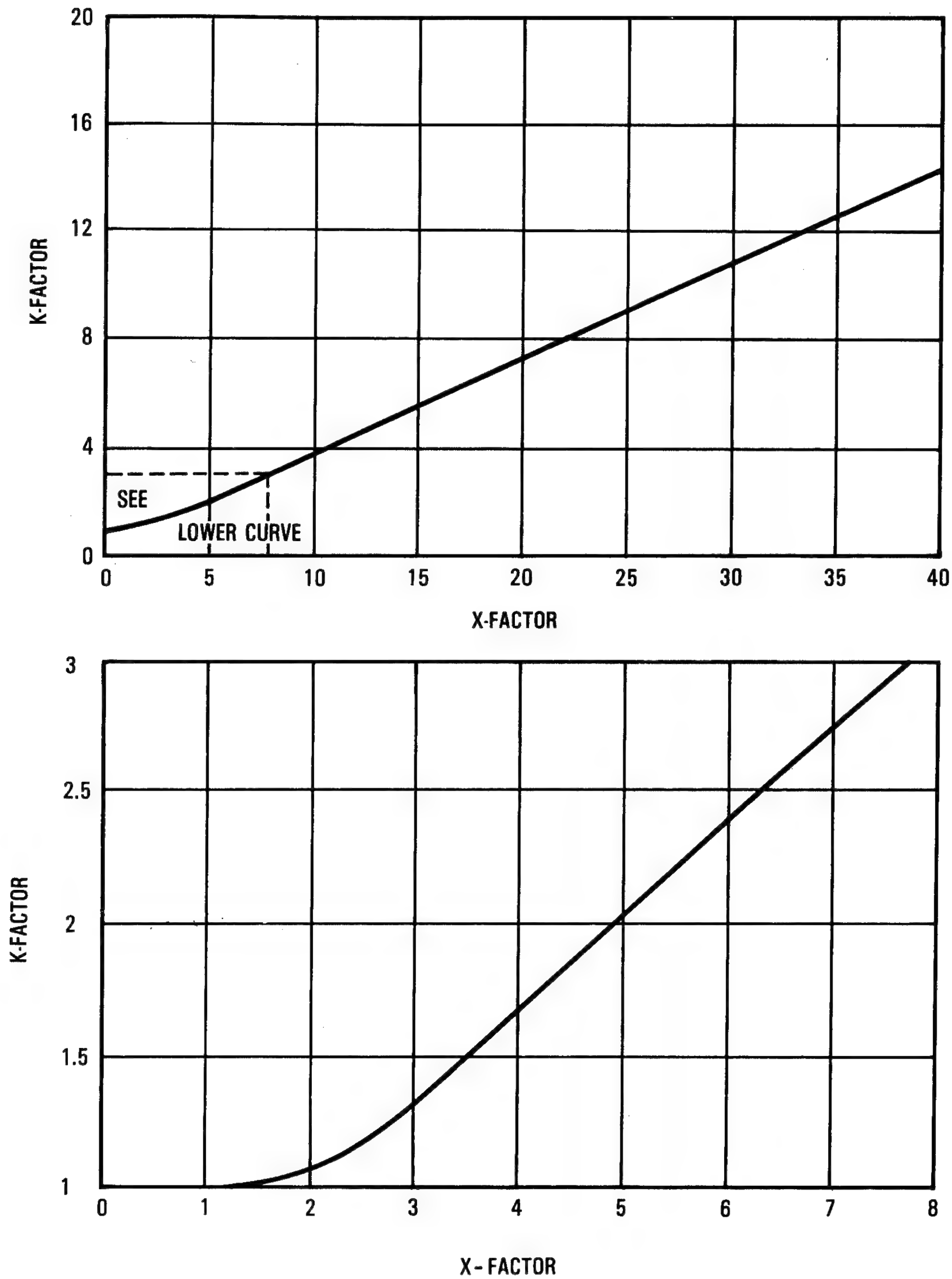


Figure 59. - Skin effect ratio, K-factor vs X-factor.
(Courtesy AiResearch)

4.2 Distribution System Analysis: Preferred System

From the foregoing discussion and evaluation of aircraft type power/feeder distribution systems, the following summary can be made:

(1) The ring type distributed bus system (Figure 53) appears to have significant merit for the IDEA/Alternate IDEA type airplanes.

(2) There are many possible variants of the ring type (and distributed radial) type systems. The proliferation of loads in the wings of the IDEA render the conventional radial systems with dedicated MELCs less favorable.

4.3 Plans And Resources For Technology Readiness

NASA is advised to pursue the development of distributed bus systems and to address particularly the development of modular assemblies that could combine load wire protection elements; these are protection modules that would preserve the reliability and integrity of the distributed bus system.

NASA should encourage the development of novel bus interface devices that could tap the distributed cables without breaking and terminating the cables.

NASA is advised that a more detailed analysis is required to address the problems highlighted in this section, namely, cable impedance, skin effect, cable power factor, cable groupings.

A new study is required to develop a compendium of cable data, pursuant to the interest in higher transmission frequencies and higher system power capacities. Multiple tables should be generated to give the R + jX values for different gage cables at different frequencies; also curves and diagrams should be developed to define the cable PF, K and X ratios, and other data.

NASA should, if high frequency power transmission at 20 kHz is projected, solicit cable manufacturers' participation in the development of special high frequency/low-loss transmission cables (coax, flat cable, concentric layer).

NASA is advised that the complexity of the power supply system is adversely affected by the number of types of systems that have to be distributed, e.g., three-phase 200 V/400 Hz, 400 V/800 Hz (or other ac type), 270 Vdc power, 28 Vdc, high frequency ac power and special voltages for digital electronics power viz 5 V, ±12 V, ±15 V, etc.

Ideally, one type of power from which all equipment might derive their unique power requirements is highly attractive when quad redundancy has also to be provided in the cable distribution system.

5. DIGITAL POWER CONTROL AND DATA DISTRIBUTION

5.1 Digital Power Control Background

With the advent of digital logic, integrated analog circuits and solid state power switching, new concepts were developed to handle greater demands for more power and more precise control of that power. The conventional power distribution/control system is severely limited by hard wiring and electromechanical switching techniques. Control logic, load switching, and circuit protection are performed via manually actuated switches with control relays and auxiliary contacts, power relays/contactors, and manually reset circuit breakers. With the exceptions of design improvements and a few special function packages, neither the basic concepts nor the application of these devices have changed in several aircraft generations.

One of the earliest of the new concepts for aircraft electric systems was designated "Solid State Electric Logic" (SOSTEL). The basic philosophy of SOSTEL was to minimize the number of power devices and the length of the power path resulting in the maximum system efficiency.

To minimize the power devices only one power controller is used between the bus and the load. The power controller provides the circuit protection as well as the load-control switching. Low level signals from the control logic are used to control the power controllers. The control logic functions are performed by microelectronic logic networks, providing significant improvements in reliability, size and weight.

Prototype solid state power controllers were developed and have evolved through the coordination of the SAE Task Group for Power Controller standards. The requirement for the prototypes indicated that some of the devices exceeded the technology of the time and the costs of advancing the technology could not be justified by the limited aircraft applications. However, by the late 1970's technology exceeded the needs and, as will be shown later, solid state power control is now cost competitive with the simpler thermal and electromagnetic components of the conventional approach. In addition, all the advantages of the new concepts for power distribution, power/load management and control processing and communications can be achieved.

Control logic methods and system concepts have also continued to evolve. There has been very little restraint since the digital technology seems to continually exceed the needs and has allowed a much faster evolution. Closely following the microelectronic logic implementations, a means to reduce the many and lengthy control wires was implemented. This new system provided multiplexing between groups of sources, the logic network, and groups of power controllers. The next phase expanded the sequential logic of the multiplex system to incorporate the control logic functions, providing for computer processing and programability of these functions. That implementation was designated the "SOSTEL II Data Handling System".

These early developments continue to form the basis of the new electric system concepts. The continuing evolution is directed at cost reduction, greater versatility,and reliability improvement. Built-in-test functions have

been expanded to cover the system from end to end including the signal sources and the power controllers. Fault detection, fault tolerance, and self diagnostic implementations have verified that digital power control can provide higher reliability, better maintainability, and lower life cycle cost than the conventional approach.

5.1.1 <u>Power control technology</u>. - The initial concepts of solid state power control (SSPC) desired a uniform current limiting action by the power controller. The quality of the distributed electric power is greatly improved by current limiting each load. With inrush and fault currents limited, the system voltage transients are greatly reduced. Current limiting was feasible for 28 Vdc controllers but slightly impractical when voltage transients and worst case conditions were imposed. For 115 Vac, uniform current limiting was barely feasible and totally impractical, considering the high energy (500 joules) that would have to be absorbed in worst case. The next stage of the SSPC evolution traded off some of the uniformity of current limit by shifting a major portion of the energy absorption to a passive component such as a wire wound resistor. This approach improved the cost effectiveness of the 28 Vdc SSPC, but at 115 Vac the additional switching devices and the higher energy absorption made such an SSPC barely practical. The evolution of the SSPC continued with advances in solid state power devices such as the bulk effect current limiter and the power field effect transistor.

The bulk effect current limiter is based on well known semiconductor physics, in that the charge carriers in a semiconductor experience a limiting velocity when subjected to high fields. The saturation of electron velocity is an ideal property for a current limiter. The factor necessary to produce a practical current limiter is a material with a sufficiently high band gap which also has a low saturation field, low net doping level, high breakdown field, and can be economically produced. Useable materials include GaAsP, AlGaAs, GaSbAs, and ZnO, all of which have high enough band gaps and exhibit carrier velocity saturation at fields in the order of 2 kV/cm. By net doping of these materials in the range of 10 charge carriers per cm^3, the power density becomes low enough to allow use of high field current limiting effects without excessive heating. Because the limiting effect is a bulk effect, the current limiter does not have hot spot problems such as occur in junction current limiters.

Since this current limit property is found in high band gap materials it is possible to operate the current limiter at high temperatures allowing large power dissipation per unit size of device. The size of the device is restricted only by material processing and packaging problems. Material processing has advanced to the point where the velocity saturation mechanism can be practically implemented to produce an economical current limiter. The use of this bulk effect current limit would greatly simplify the power and the control circuitry. The ac module could use standard SCR technology with its well known characteristics and reliability. The BIT circuitry would also be simplified as the amount circuitry to check would be reduced. These simplifications further improve the reliability and cost effectiveness of the solid state power controller.

The Power FET has had a rapid expansion in technology and application since 1979. Ratings of present devices exceed 1000 volts and can efficiently switch over to 300 watts per device. The Power FET offers some unique advantages over power transistors. These advantages are:

- High Input Impedance: These devices can be driven easily by standard CMOS gates. This property will greatly reduce the control circuit power dissipation raising the overall efficiency.

- No Secondary Breakdown: The basic mechanism of FET switching does not involve the transistor characteristic of secondary breakdown. This characteristic allows the full utilization of the devices dissipation and voltage ratings.

- Can be Paralleled with Ease: Because these devices exhibit a positive temperature coefficient, they can be paralleled without current hogging.

- Can be Seriesed with Ease: The high off impedance allows a series connection to be conveniently implemented for higher voltage application.

- Bulk Mode Control of current: This basic property permits the voltage drop of these devices to be reduced to very low levels by paralleling many devices. Other types of power switches exhibit junction voltage drops that set the low limit of power loss.

- Low Power Density: The fact that these devices can be paralleled allows the power that is dissipated to be spread over a larger area which will increase the surge capability of the system.

- High Speed Turn-Off: The FET is a majority carrier conductor therefore no recombination delay is incurred. This eliminates current overshoot and reduces power dissipation on trip out.

These advantages have significantly improved the cost effectiveness of SSPC's, particularly for high voltage requirements. Also the high speed of the power FET allows application of instant trip which effectively limits fault current without high energy absorption. Normal inrush current would pass through but should not create significant voltage transients. With the instant trip approach, the only significant energy absorption required of the SSPC is for the inductive energy stored in the load.

The level of instant trip is subject to trade-offs. A level set at 15 times rated current would be required for universal application. For the incandescent type loads that require that level there is little problem. However, if a primarily inductive overload were allowed to charge up to that level, which is more than twice the normal inductive inrush, the energy that must be absorbed by the switching device then would be greater than 4X the normal maximum. Thus a preferred trade-off would be to set the instantaneous

trip at 7.5X. The incandescent type load would then require a 2X rated circuit, which may already be required due to size or voltage drop considerations. Another consideration is that with zero voltage turn on, the inrush of incandescent loads may be limited to less than 7.5X.

The digital interface to the power controller has also evolved through the years resulting in several options. The controller was originally conceived as a stand alone, relay type component which was wired into a power system. New applications will still require some directly controlled solid state or hybrid power controllers. However, for reduced cost and cable complexity SSPC's will be applied as an element of a system - a card module in the load management center (LMC).

As a stand alone or a system element, the SSPC has "smarts" of its own, performing several functions. Its relay function requires that it provide an auxiliary or interlock signal which is an ON-OFF logic/status signal. Its circuit breaker function requires that it provides a trip-free function with memory to protect itself as well as the load circuit. A trip status signal is also required. Also, since its "mechanisms" are solid state, there is a need for built-in-test (BIT). The interface to the SSPC thus requires command signals of ON, OFF, Reset and Test, and status signals ON, OFF, Trip and Fault. For interface, eight signals could be provided, four command and four status. Simple binary conversion reduces that to two each. A functional combination of OFF and Reset along with pulse timing provides all command requirements with one signal. A continuous "low" equals OFF & Reset, continuous "high equals ON & Test, a short "high" equals OFF & Test, and a short low is ignored. Thus three digital signals can provide the required interface. A further refinement, proposed by NADC, is known as the BIT interface. The BIT interface combines analog impedance along with the analog timing and combines the status signaling on the same line as the command signaling.

The interfacing of the SSPC as a system element of the Load Management Center (LMC) allows further options. As suggested in a recent Electrical Multiplex (EMUX) system study, some of the SSPC intelligence could be transferred to a common processor in the LMC. However, a more detailed analysis of that concept indicates a decisive disadvantage in circuit isolations, complexity and overload response time. Circuit isolation is required for voltage level shifting, avoiding ground loops and electrical noise decoupling. Current sensing over the wide range required (150:1) requires tight coupling to the load circuit which is contrary to the electrical noise decoupling required. Also, the wide ranging, low level current signal is more difficult to handle than discrete logic signals and circuit complexity would be increased in the sampled data system. The built-in-test and the overload calculations are relatively simple and are intimately associated with the isolated power element. Separating these functions would require more isolation and selection circuitry than the circuits saved through shared calculations. Also redundancy would be required in the shared circuitry. Overload response time would be jeopardized particularly at high overloads, so to minimize the power dissipation and the cost of the power element the instant trip response must occur in microseconds. An accurate scan of 64 to 128 current sensors would require at

least 0.5 seconds, affecting the accuracy of overload trip at the 300% level. Therefore it is recommended that the SSPC functions be integral and self contained.

Failure effects must also be considered in the interface to the SSPC's as well as in the LMC system. Aircraft power distribution systems are uniquely different from industrial or commercial power distribution systems. Rather than just branch circuit protection, overload protection is provided for individual load circuits. This results in the need for hundreds of circuit breakers. This individual protection provides an effective fault isolation so that a single load circuit failure will not effect other circuits. The SSPC provides the individual protection of the load circuits, but a fault in any shared SSPC control circuit would disable all of the sharing group. The interface to the SSPC then must minimize fault propagation. Therefore any shared circuitry must be redundant or fault tolerant. This fault isolation requirement extends to the LMC processor as well. Since the processor shares control of a hundred SSPCs it must be made redundant or fault tolerant.

5.1.2 Power controller developments. - Over the past 15 years hundreds of remote power controllers of various types and ratings have been developed and prototyped. This technology is now considered off-the-shelf by those cognizant of advanced aircraft electric system technologies. Additional efforts are being made to refine and improve performance and life cycle cost of these components. The development of technology for solid state power control has been strongly supported by NASA Lewis Research Center (reference 5). They have been sponsoring the development of power semiconductor devices, achieving high current and high voltage capabilities. NASA LeRC has also been developing remote power controllers for space applications, beginning with the 28 Vdc units in the Space Shuttle, and recently achieved 1000 Vdc with 25 kW capability.

The most recent solid state power controller (SSPC) developments have been directed at the load management center (LMC) concepts, wherein the power controller is a card module incorporated within the LMC unit. One approach, configured for ARINC 600 box specifications, is proposed and specified in the AAES Control Technology Demonstrator (CTD) design for the Air Force Wright Aeronautical Labs (AFWAL) (reference 6). The specified configuration is shown in figure 60. Another approach employs the improved standard electronic module (ISEM) and is being employed in the advanced aircraft electric system "hot bench" at the Naval Air Development Center (NADC) (figure 61).

The SSPC is most cost effective at the lower current ratings. As shown in the "Load Distribution Comparison" (table 10), the majority of the loads (76 percent to 92 percent) are within the present cost effective range of solid state. The higher current loads are handled by the hybrid power controllers consisting of solid state control and protection circuitry operating a relay for the high current switching. The hybrid power controller is usually a larger stand-alone component which replaces the power relays and eliminates the need for an accessible power circuit breaker. The hybrid power controller has not received the same standardization effort as the SSPC; however, various implementations of the hybrid approach have been applied to a limited extent in the majority of current aircraft (references 7 and 8).

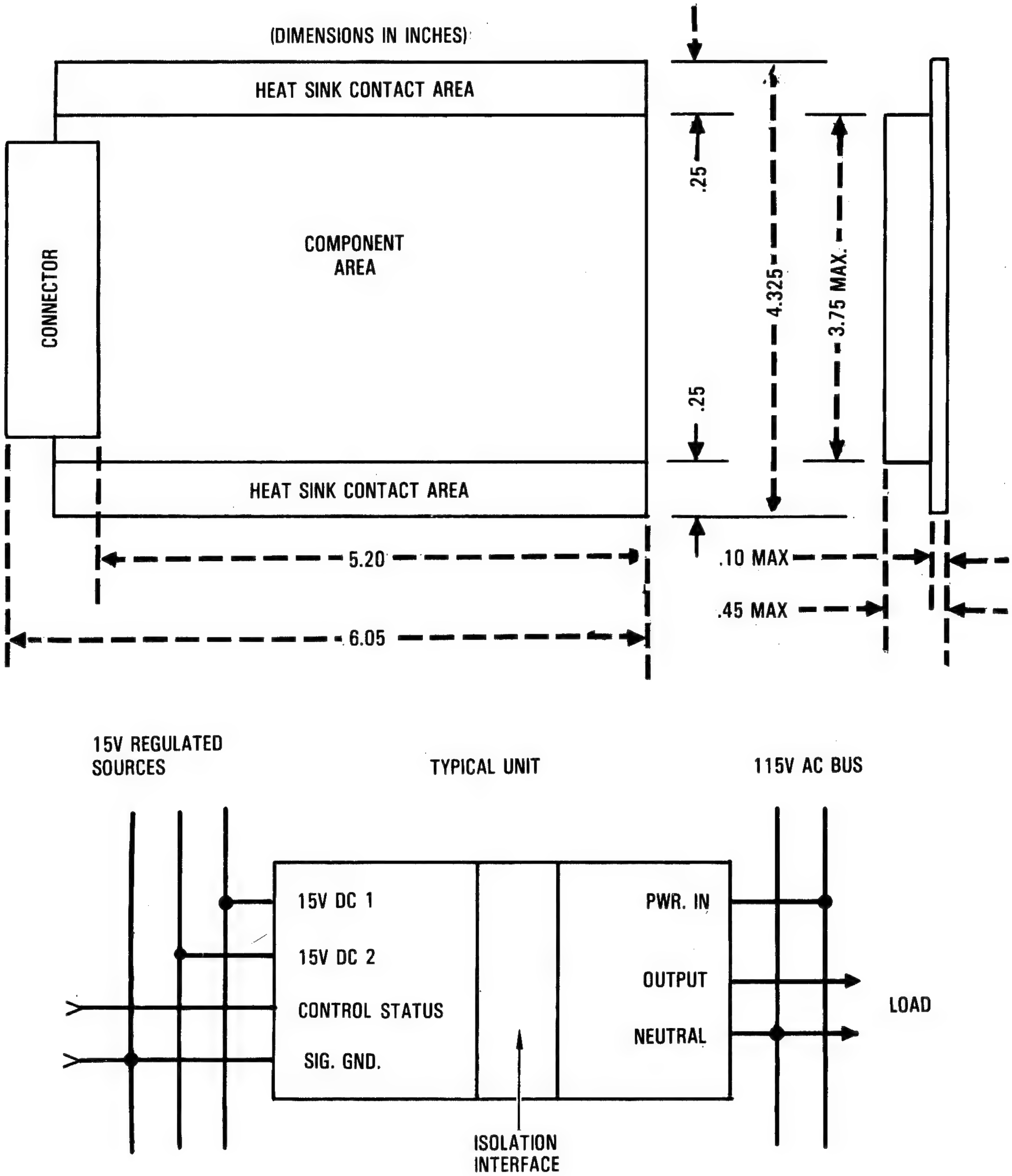
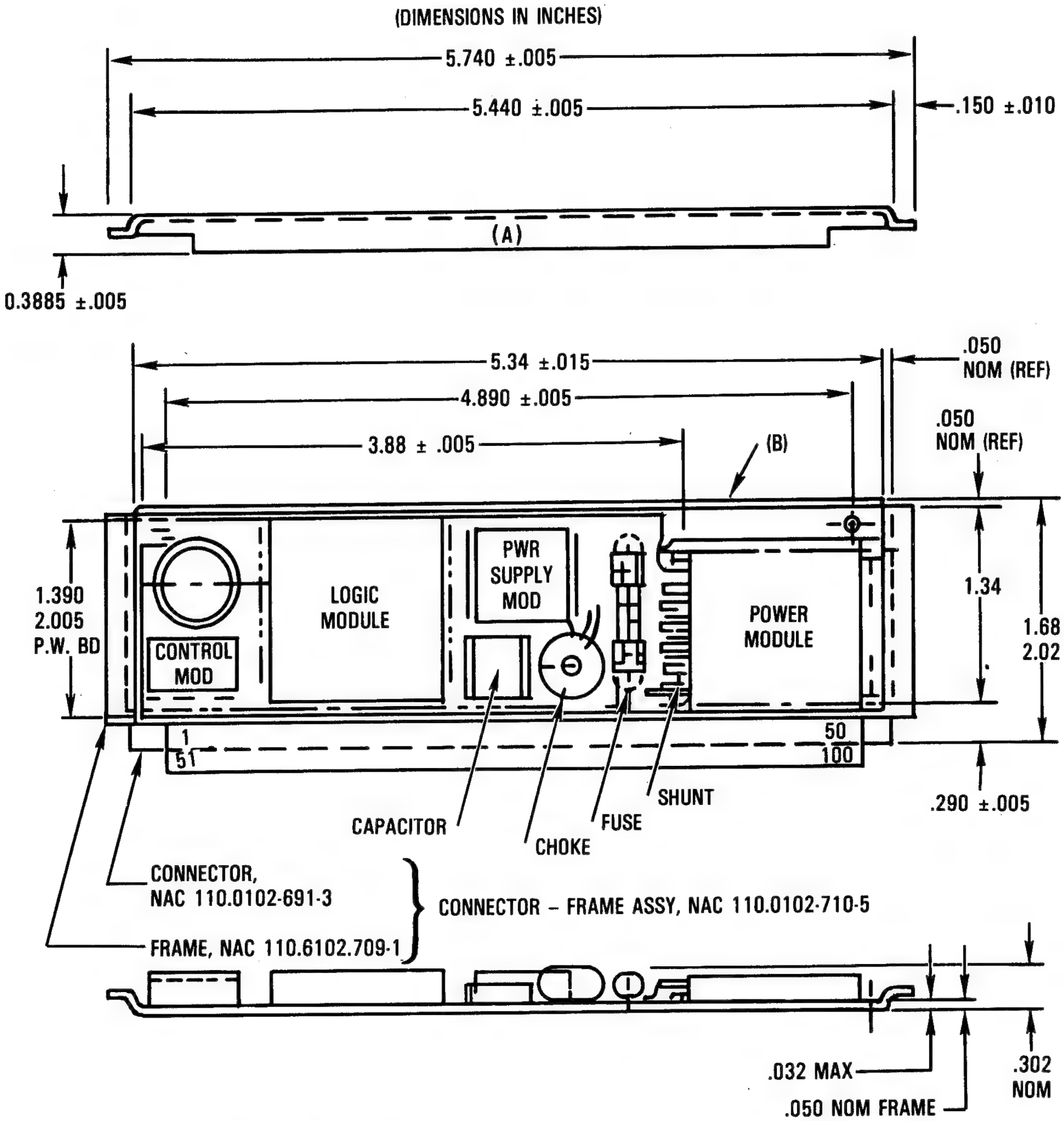


Figure 60. - Power controller, 115V 400 Hz load switching SPST configuration.



CONNECTOR NO. – NAC0102-691-3

PINOUT:

FUNCTION	PIN NO.(S)
CONTROL	2
CONTROL GND	4
POWER IN	37, 38, 87, 88
POWER OUT	34, 83, 84

NOTES: (A) NAMEPLATE LOCATION
(B) COMPONENT TYPES AND LOCATIONS ARE FOR ILLUSTRATION ONLY

Figure 61. – Power controller, 270 Vdc, load switching SPST, normally open, 1, 2, and 5 amperes.

TABLE 10. - LOAD DISTRIBUTION COMPARISON

Aircraft Type Generator Capacity	C-15 120 kVA	A-7 25 kVA	C-X 60 kVA	CTD 120 kVA
Total Loads				
Ac	260	138	400	240
Dc	253	203	--	260
Ac Distribution - Cumulative				
0.5A	13%	44%	56%	--
1.6A	41	(1A)63	(1A)75	--
2.0A	48	74	(2.5A)90	52%
3.0A	57	--	--	64
5.0A	64	87	95	74
7.5A	76	--	--	84
10.0A	90	100	100	91
15.0A	94	--	--	95
Other	100	--	--	100
Dc Distribution - Cumulative				
0.5A	9%	30%	--	--
1.6A	66	49	--	--
2.0A	75	61	--	75%
3.0A	81	--	--	87
5.0A	88	81	--	91
7.5A	90	--	--	95
10.0A	92	99	--	98
15.0A	96	100	--	99
Other	100	--	--	100

5.1.3 Power controller applications. - The application of remote power control in production aircraft has been surprisingly limited. An early standardized product was the remote controlled circuit breaker (RCCB). This hybrid component was developed to meet the needs of the wide body aircraft. The RCCB provided a substantial weight reduction by eliminating many power cables between the main electrical load center and the manual circuit breakers in the flight station. The RCCB and other customized hybrid power controllers have been included in most subsequent aircraft designs. Even smaller aircraft such as the BAe-146, the SF-340, the F-16, and the AH-1S helicopter have advantageously employed them. The RCCB was also applied in the Space Shuttle test craft and custom hermetically sealed hybrid power controllers were developed. The Shuttle design incorporates an EMUX system utilizing hundreds of SSPCs, thus providing the technology and effectiveness of remote power control.

To date the most extensive application of remote power control is in the B-1. A complete and separate EMUX system is incorporated into this advanced

aircraft design. This EMUX system employs more than 1000 remote power controllers resulting in the elimination of 57 miles of control and power wiring and a weight reduction of 1200 pounds. However, the remote power controllers in the initial aircraft are a hybrid type and cannot be reset remotely. A set of SSPC's were justified and designed to be applied in the fourth test aircraft but were not because of the cut-off. The reinstated B-1B continued with the initial designs but could be converted to solid state in the near future.

5.2 Digital Data Multiplexing

Because of the growth of thousands of feet of different wire segments in advanced military and commercial aircraft, industry has looked to the application of digital multiplexing (MUX) concepts as the technology for reducing the wire quantity/weight and bringing viability to the aircraft's electric/avionic system.

Basically the philosophy of multiplexing is predicated on the use of a few data-transmission cables over which many hundred of thousands of signals may be transmitted from the flight station/computer control center to the remote loads (avionic equipment or load controllers). The technology of multiplexing is by no means new and it is used widely in many industrial computer systems and in aircraft.

Up to this time, in the commercial aircraft electric systems, multiplexing has been constrained to the passenger service call systems and the passenger audio-entertainment systems. The latter system is sophisticated in that it involves the transmission of high-quality audio signals, covering a frequency range of 15 to 40 kHz. There are fifteen separate channels, each using 12 bit resolution of audio data to minimize the quantization noise and increase the dynamic range. Each channel is updated at a rate of 25,000 sec and the serial clock rate is 4.5 MHz.

In flight service these MUX systems have not been without their problems, but the development status of multiplexing can be said to be at a higher level of development than those early Passenger Entertainment/Service (PE/S) systems. Unlike any audio-entertainment systems, an electrical multiplex (EMUX) system will be involved primarily with the transmission of discrete control signal data, rather than the transmission of multi-bit resolved analog data. However, for electrical quality and usage monitoring and utility service for other systems the EMUX system would be involved with some analog digital conversion.

The remote loads in an EMUX-configured airplane will still use separate, dedicated power wiring so any wire/quantity/weight reduction will therefore come mainly from the elimination of the lighter gage control wiring and a shortening of the power wiring in the aircraft. However, it is implicit in this, that if major benefits are to be derived from an EMUX system, it must be achieved by a complete revision of the conventional power bus architecture now prevalent in current aircraft. Simplification of the power bus architecture is in fact one of the primary objectives of advanced power management and it

can only be achieved if buses are located in close proximity to the electrical load moments in the aircraft.

Figure 62 is a simple schematic of an idealized layout of an advanced power control system in which buses are located in the flight station, wing-roots, empennage, and other load areas in the aircraft. To this extent, it is a "distributed radial" system, as opposed to "centralized radial" distribution, still used in many past and current aircraft systems. In the latter system, as exemplified by the early jet/prop jet aircraft, the MELC was usually located in the flight station area or in a forward fuselage area. As a result, the control wiring lengths were short and the power-wiring lengths were long. This is antithetic to the objective of an AAES system. It is therefore clear that if weight reduction is a main criterion, it can only be achieved if the power distribution is such that the electrical load centers are distributed and close to the loads.

However, most of the earlier multiplexing activity was in the avionic area where such systems as digital air data systems (DADS) were developed into systems such as digital data acquisition system. Multiple interior communications system (MINCOMS) was a very early (1964) Navy sponsored program for a

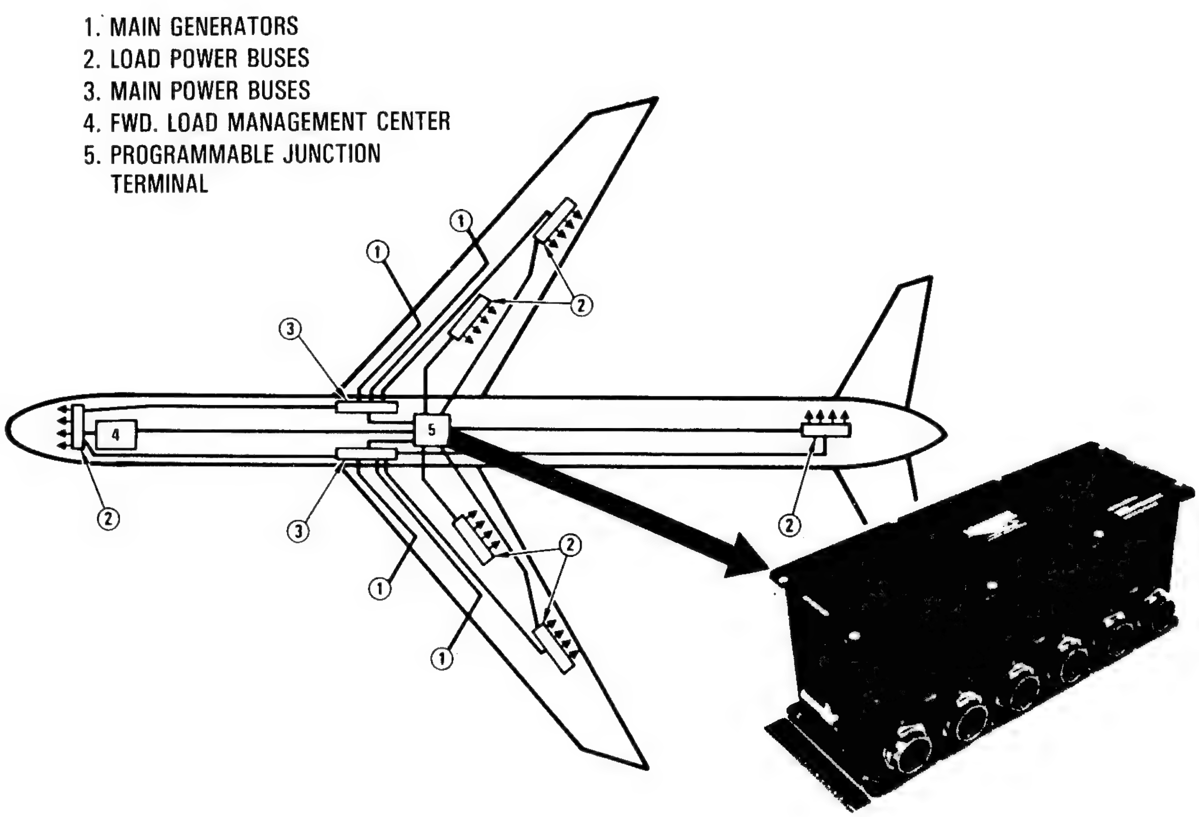


Figure 62. - Advanced power control system.

modern aircraft communications system and it utilized a combination FDM/TDM signal transmission system. This study, conducted by RCA for the Lockheed P-3, showed that some 34 coaxial cables could replace approximately 4000 wires in the conventionally wired airplane. Weight savings up to 1200 lb were predicted for the MINCOMS.

There is ample experience in military multiplexing systems, as exemplified by those used in the B-1, F-15, F-16, F-18, and Lockheed's S-3, P-3, C-141, and C-5 aircraft. For the central integrated test systems (CITS), aircraft integrated data systems (AIDS), and other data systems, multiplexing has been a valuable adjunct to the aircraft's electric systems since it eliminated many thousands of feet of wire which would have been required for the status and instrumentation monitoring. The selection of extensive multiplexing in the B-1 bomber and the more recent selection and expansion of AMUX in all new fighter aircraft, stand as endorsements of the aviation industry's commitment to the utilization of this new technology in modern weapon systems.

5.2.1 Multiplexing technology. - Multiplexing is data transmission in which signals are transmitted simultaneously over frequency-assigned channels frequency division multiplexing (FDM); or sequentially as time-division multiplexing (TDM); or a combination of both, as an FDM/TDM system. The signals themselves may be transmitted as analog data or digital data, depending upon the type of modulation technique. Analog data transmission has the merits of simplicity in that no A-D (analog-to-digital), and no D-A (digital-to-analog) converters are required. However, for noise immunity, and other reasons, industry has standardized on pulse-code-modulated (PCM) systems in which digital data (words) are transmitted. The data words could be transmitted on paralleled lines (parallel PCM) or sequentially on fewer lines (serial PCM).

The development of a standard for serial PCM transmission began in early 1968 with the formation of an SAE subcommittee on "Multiplexing for Aircraft" (SAE-A2K). The committee developed the first draft of a digital time-division multiplex data bus standard and expanded that into a full system specification "Control Group, Electric Power" MIL-P-81883. The developing document represented the best that industry and military engineers could define and provided a sounding board for evolving ideas. It also provided the basis for USAF MIL-STD-1553, first issued in 1973 and now a uniform world standard for "Aircraft Internal Time Division Command/Response Multiplex Data Bus". Since the inception of the SAE-A2K, the industry has been designing and producing hardware for various MUX systems. Some of these systems were developed during the standardization era (i.e., B-1 and F-15) and are similar but incompatible with MIL-STD-1553 which was first applied on the F-16. Since then several modest revisions have occurred through coordination of industry and military in the SAE-A2K and via symposia, contracted studies and development programs. Recently the subcommittee was upgraded and renamed committee on "Aerospace Avionics Equipment and Integration" (SAE-AE9). As the name implies its scope has expanded, continuing development of future MUX standards, while adding a microcomputer and software subcommittee. An EMUX subcommittee was also established to promote, specify and standardize architecture and components for aircraft electric systems.

The key characteristics of the MIL-STD-1553 data bus are based on the requirements for a party line bus with up to 32 taps to operate asynchronously in a command-response (master-slave) duplex mode. The signals are transferred over the data bus in serial digital pulse code modulation form, the transmission bit rate on the bus is 1.0 megabits per second, and the data code is Manchester II biphase level. Each data word consists of a unique sync waveform (3 bit times), sixteen bits of data, and a parity check bit, as shown in figure 63. An intermessage gap and response time are also specified. The specification also covers cable and cable impedance, attenuation, termination, cable taps and stubs, and terminal input and output characteristics (waveform, noise, symmetry, common mode, and impedance).

The rapid growth of large scale integration (LSI)/very large scale integration (VLSI) and the availability of integrated circuit (IC) MUX chips has led to the development of a multiplex capability that can meet the most sophisticated requirements of the commercial and military airplane. Thus, many alternative MUX designs have been and are continuing to be developed.

For commercial aircraft, the leading standard is ARINC specification 429 applied to information transfer between avionics equipments. The specification is for point-to-point, unidirectional, moderate bit rate data transfer. While this is suitable for transferring a large quantity of data to one or several equipments, it is not usable for the widely scattered, discrete input-output linkages required in the electric system. For industrial controls there are as many MUX designs as there are product vendors, and basically these designs are permutations and combinations of each other providing a variety of parameter trade-offs, with little if any overall advantage.

Because of the characteristics of the MUX signal transmission format all signals must be conditioned and digitized before inputting onto the data bus. Amplitude scaling of analog data into the desired number of bits is necessary. Typically in most MUX systems, conditioning, buffering and formatting is accomplished inside the equipment, or in remote terminal units (RT). Figure 64 shows a very simple block schematic of a MUX element in which an RT performs the role of interfacing the individual equipments with the data bus. That approach lends itself to retroactive MUX installations. It proliferates the use of interface connectors however and there is a large amount of intra-wiring between the equipments and the individual RTs. The alternative and better approach is where the intrawiring and additional connectors are eliminated by having bus interface electronics built into each equipment.

Another example of system architectures is that shown in figure 65. These schematics were prepared by Vought-LTV as a part of the work on the A-7 simulator program and illustrate the proliferation of wiring and connectors when the multiplexer data terminal (MDT) is a stand-alone type (figure 65a) as opposed to the configuration in figure 65b where the MDT is integrated into the load management center (LMC). The latter permits intrawiring and interfacing with the power controllers to be accomplished inside the LMC. This approach offers potentially higher reliability and eliminates many connectors.

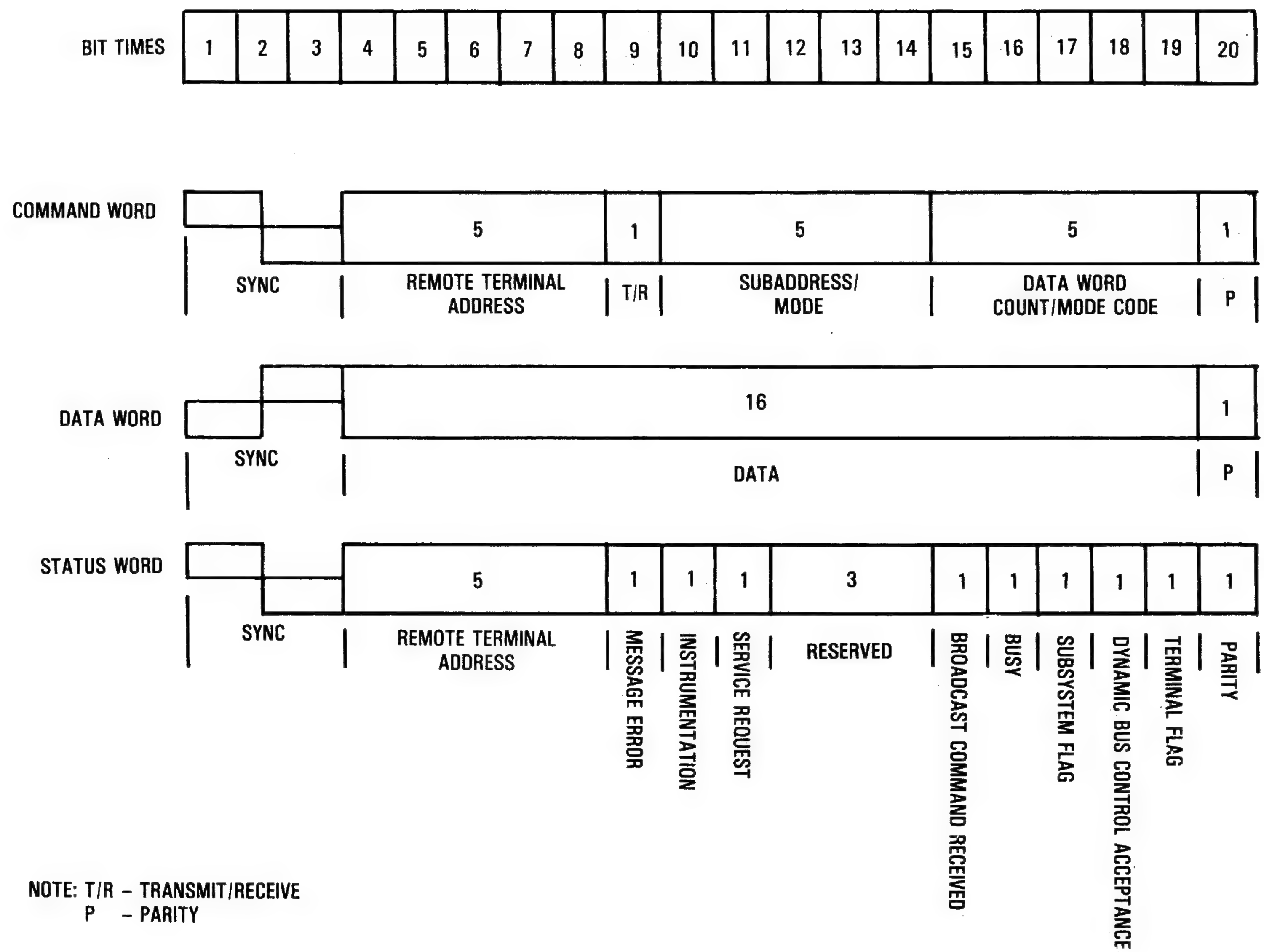


Figure 63. - MIL-STD-1553 word formats.

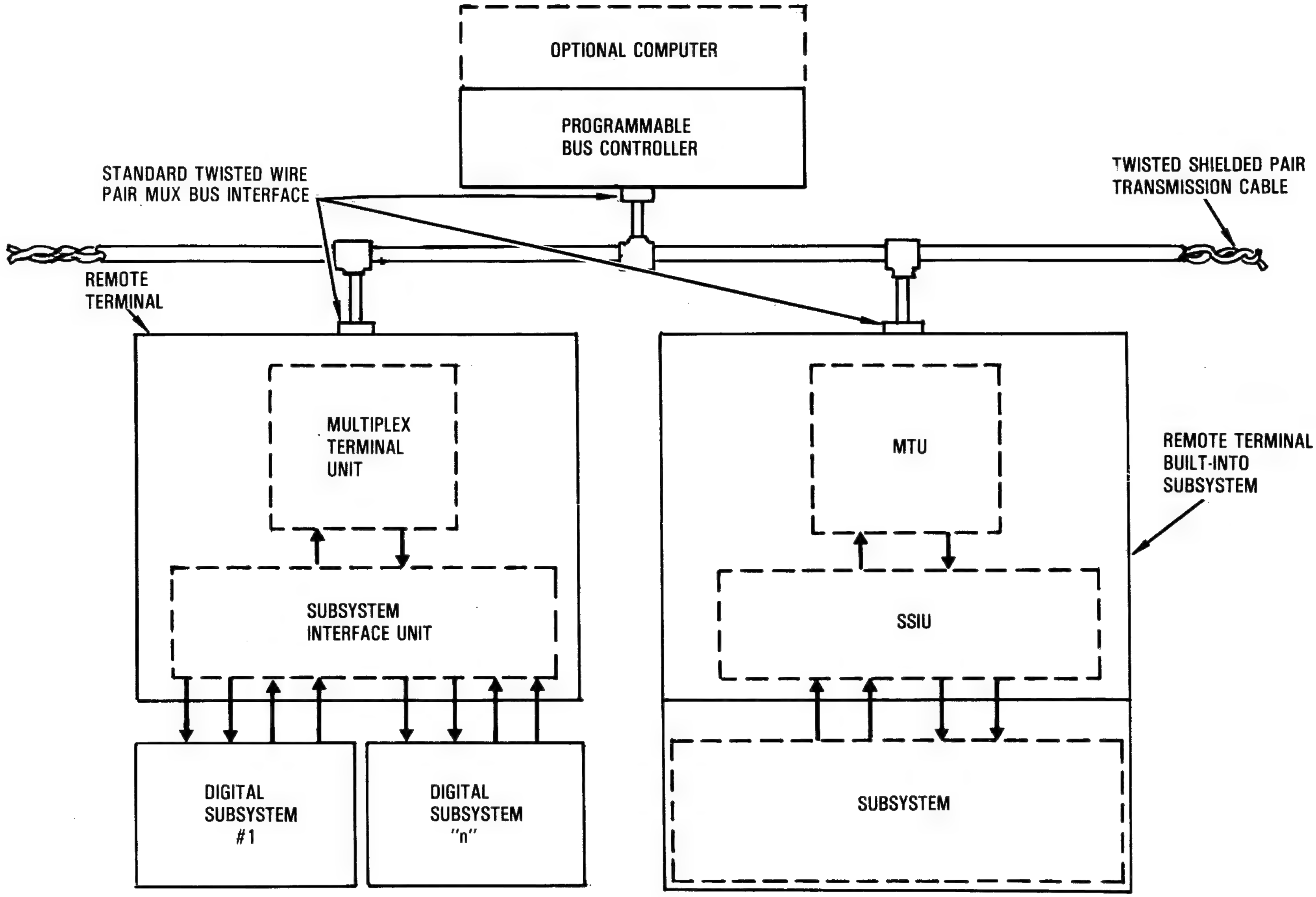
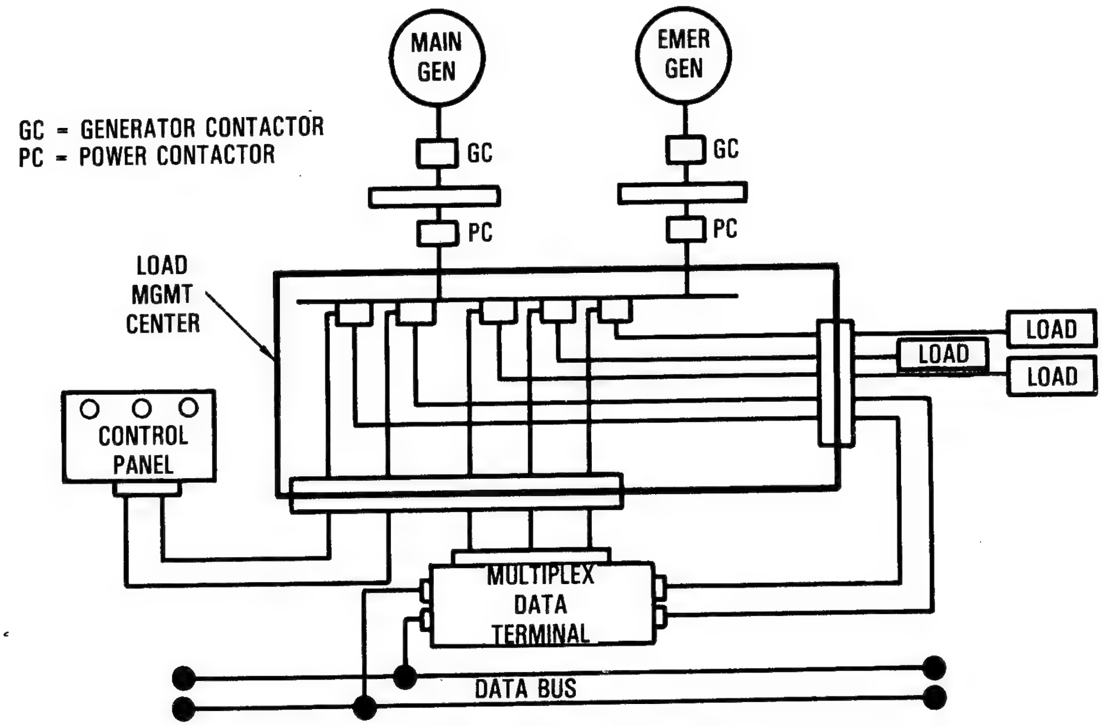
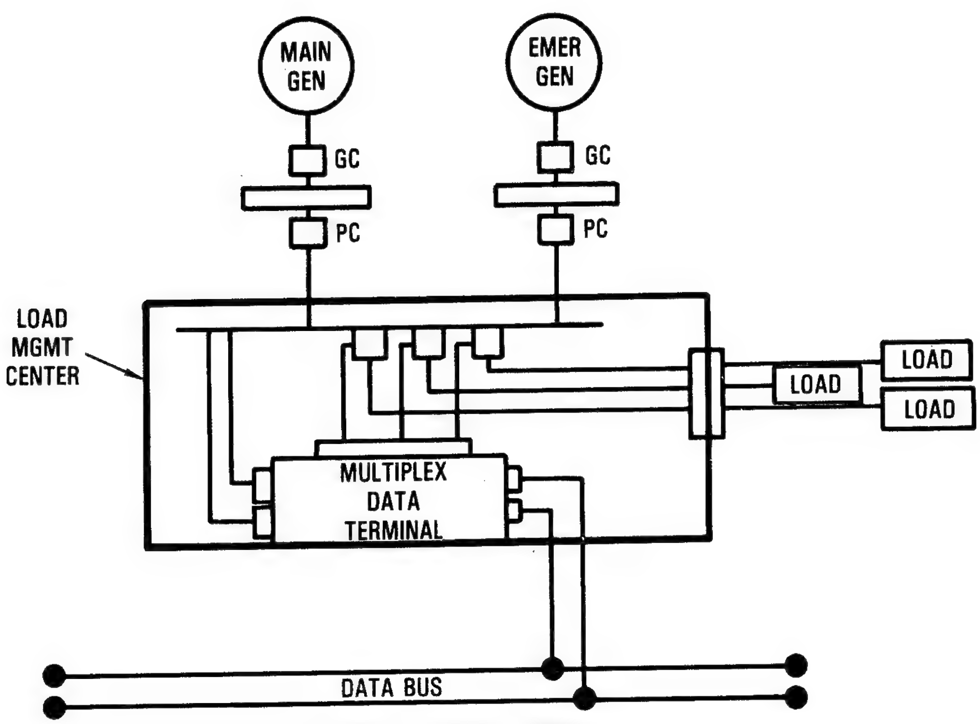


Figure 64. - Elemental bus architecture.



(a) TERMINAL OUTSIDE CENTER



(b) TERMINAL INSIDE CENTER

Figure 65. - Load management center design concepts. (Courtesy Vought-LTV).

5.2.2 Multiplexing Applications. - Because of the success in the developments of multiplex components and systems, there is an increasing commitment by the aerospace industry to multiplexing. The selection of multiplexing for modern weapon systems such as the B-1 and F-16 airplanes endorses the military's confidence in the utilization of multiplexing for major roles in these aircraft (reference 9).

The Air Force's promotion and technical direction of extensive multiplex utilization for the EMUX, AMUX and CITs system in the B-1 airplane was itself based on the critical space and weight problems in this aircraft and the need to incorporate more sophisticated management of the aircraft's avionic/weapon system and the electric system. In F-16 airplanes nearly all the major weapon systems are interfaced via a MIL-STD-1553 data bus to the fire control computer and a back-up computer in the internal navigation unit (figure 66). The AMUX system for the F-16 was selected not so much for any wire weight saving, but rather for the versatility and flexibility, offered by multiplexing.

In a nonairborne application, the U.S. Army Tank Automotive Research and Development Command installed a MUX system in the M60A2 tank on the basis of an analysis that indicated improvements in the inventory logistics and life cycle costs of the tank. The U.S. Army has additionally selected multiplexing for LAMPS and for the Advanced Attack Helicopter. There is therefore no lack of development and implementation experience in multiplexing in military systems today.

5.3 Digital Electric Management Evolution

The aerospace electrical industry has proposed methods of automatic load management that would provide significantly better management and improve data display and monitoring methodology for advanced aircraft (ref. 10). The AAES

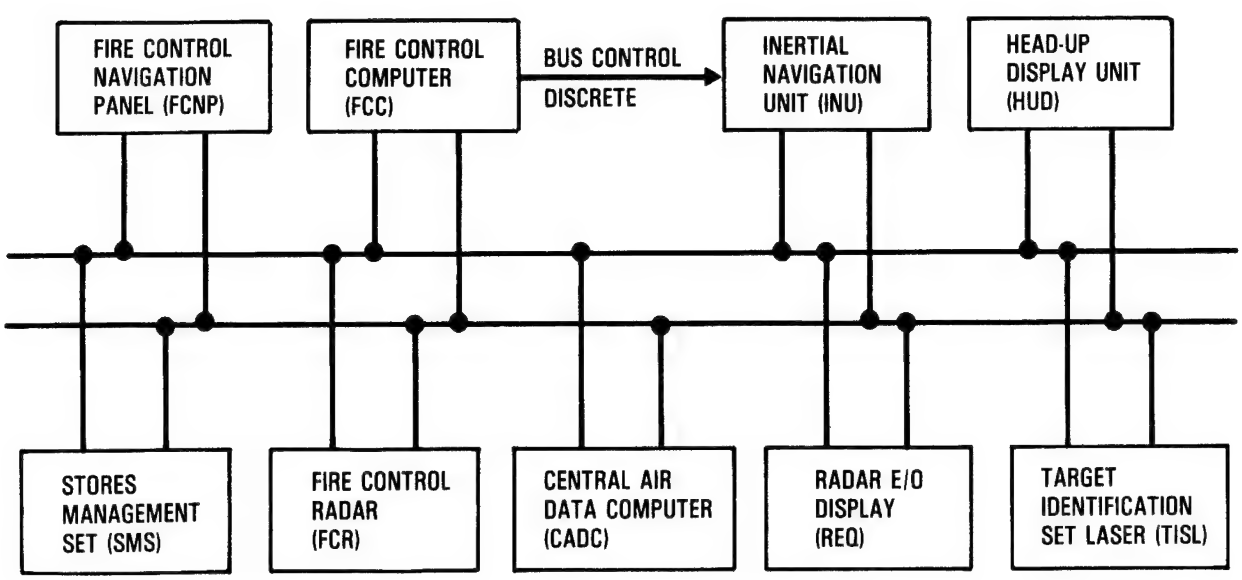


Figure 66. - F-16 avionics system architecture.

designation is the most current and more encompassing than EMUX used in preceding sections, although the acronyms are often used interchangeably.

The initial activity, already mentioned, was the NADC/AFWAL sponsorship of SOSTEL and Data Handing System developments for the A-7 airplane. The purpose and precepts of the A-7 simulation program were to make a direct comparison between a solid state/multiplexed electrical system and a conventional hardwired A-7 electric system. In the course of this program the synergism of solid state power controllers, solid state signal sources, programmable processor control, and a data handling system (multiplexing), was realized.

The data handling system for the A-7 simulator comprised the following:

- A Master Control Unit (MCU)
- Six Remote Output Terminals (ROTs)
- Six Remote Input Terminals (RITs)
- Fault-Indicator Panel (FIP)
- Pilot Control Panel (PCP)

Each of the RITs and ROTs had a 64 channel capability. The RITs were designed for interfacing with solid state signal sources (SS) and the ROTs were designed for interface with solid state power controllers (SSPCs).

In the evolution of that system the function of the RIT and ROT were combined into a universal remote terminal. Built-in-test (BIT) was incorporated and extended into the SS and SSPC. The MUX was also changed to conform to MIL-STD-1553B.

Another more advanced application of a multiplexed electric system is the EMUX system in the B-1 bomber. Here the objectives were to remove some 57 miles of wire from the airplane and to provide an advanced management control system that would reduce the crew workload in the military environment. The other objectives were to reduce life cycle costs and bring viability to the aircraft wiring installation.

The B-1 EMUX system (figure 67) is designed to handle some 9000 signal inputs/outputs, which are handled by 22 to 26 multiplex units (MX) and expandable to 42 MXs (21 per side). Two dual-redundant data bus systems are routed separately in the fuselage and these interface with the MX units via couplers which protect the data buses against catastrophic faults in the RTs. The heart of the EMUX systems is the centralized Boolean control processors which solve the equational circuit logic and supervise the control of the electrical system.

The B-1 EMUX system has a 210K bit programmable memory (PROM) which permits a wiring reprogrammability as required by circuit modifications and/or equipment updates. In fact, the technology of multiplexing/processor design is best exemplified by the fact that each B-1 control processor can solve

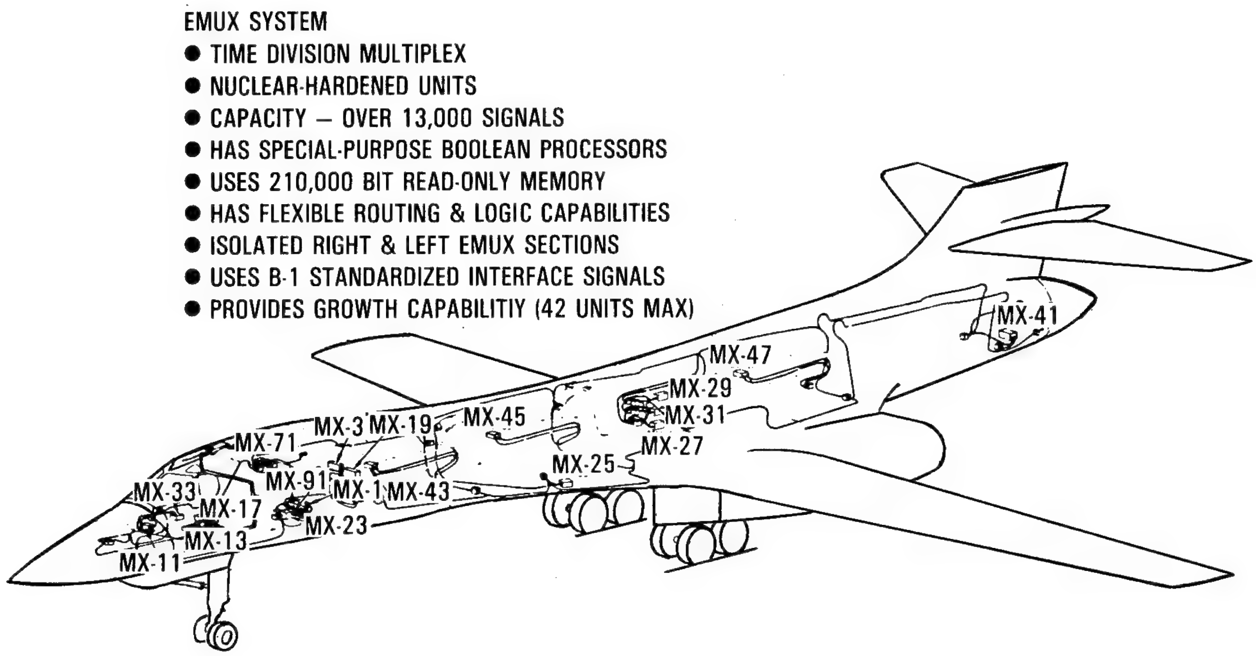


Figure 67. - B-1 EMUX system.

5800 expressions with 28,000 literals in less than 30 ms. Further, to meet the Air Force's stringent requirements, all the EMUX hardwire for the B-1 aircraft is designed to meet full electromagnetic interference/nuclear hardening protection.

5.3.1 Advanced Aircraft Electrical System Technology. - The conventional aircraft electric system consists of a multitude of individual circuits, typically including a control circuit, and a load circuit. The control circuit consists of a circuit breaker, control switches, relay contact logic, and a power relay. The load circuit, usually fed through several stages of bus and feeder breakers, consists of a load circuit breaker, power switches and relay contacts. This conventional scheme requires thousands of components and miles of wire, however it does provide the control required. Also the individual circuits and circuit breakers, if properly coordinated, maximize fault isolation. The goals of AAES are: to minimize wiring, both control and power; to reduce components, combining control and protection; to improve the quality and availability of electric power; and to maximize the viability of the system, accommodating the rising rate of modifications and additions. These same objectives tend to reduce fault isolation therefore a major requirement in AAES is the incorporation of redundancy, fault tolerance, and built-in-test to restore the level of fault isolation. The developments and applications described in the following section have proven that this reliability requirement can be met and improved.

The heart of the AAES technology is the remote power controller. As described in the previous section (Digital Power Control) the capability for remote control provides a major weight saving while improving the logistics of

power distribution. This advantage has been exploited, to a very limited extent, in all current aircraft, but the full realization of its potential requires a revision of the total system, the AAES approach.

The load management center (LMC) is an essential block of the AAES which includes the remote power controllers. As a "black box", an LMC is merely interposed between the power sources and a multitude of loads with a connection for control information. The technology of remote power control has kept pace with this system concept through the development of the module configurations shown in the previous section. The LMC concept is also a logical expansion of the remote terminal required to multiplex the control information, thus joining the multiplex technology with the remote power control technology in the same "black box".

The power system processor (PSP) is the brain of the AAES, generating the control information through the solution of system control equations. The technology of the PSP has also kept pace with the digital electronic evolution starting with discrete combinational logic, then special purpose sequential logic, and general purpose minicomputers (AN/AYK-14 and MIL-STD-1750). These advances in control processing provided additional advantages to remote power control and AAES. The sequential logic was compatible with multiplexing which replaced miles of wire required to distribute control information throughout the aircraft. The sequential processing also provided for software programmability which greatly increased the viability of the system. The minicomputer improved the programmability, and increased functional capability, thus improving the cost effectiveness.

This increased functionability is applied to power/load management wherein load priorities are dynamically allocated on an individual basis thus providing maximum utilization of the generated power, i.e., reducing the need for excess generating capacity. With the individual load management capabilities of AAES the hardware partitioning of the electrical distribution system (emergency bus, essential bus, etc.) is not required. This allows consideration of alternative power distribution architectures, concentrating on uniformity and integrity, such as the redundant ring main architecture.

This automatic power/load management also reduces crew work load. This is further reduced by the diagnostic capabilities of the computer managed AAES providing: transient filtering (auto reset, etc.), prioritized warnings (display by exception), and fault recording (nonvolatile memory). Crew work load is also reduced through integration of electric system warning and status display with the flight system displays.

The centralized processing of control information is consistent with these expanding functionality requirements, providing maximum integration and coordination throughout the aircraft. The command/response protocol of the 1553 data bus is equally compatible. However, the processing requirements are growing more rapidly than central computer technology. For a large aircraft such as the B-1, the processing requirements had to be split between two sets of special purpose processors. Development of new software methods can improve processing efficiency and system response times. One method, employed in the NADC development, applies a dynamic scheduling algorithm that responds

only to state changes, thus solving only the control equations that are affected by the state change, instead of solving all equations all the time. Another method, employed in the AFWAL development, partitions the system according to individual response requirements and schedules the processing accordingly.

The greatly expanded application of electric equipment in future aircraft as proposed in this study could result in a large increase in the quantity of data that must be distributed. This can be handled with multiple data buses or an increase in the speed of the data bus, either of which add cost and complexity. A better approach, as proposed in the AFWAL study, is to add intelligence to the load management centers (LMCs). This allows distributed parallel processing as well as data reduction. A microprocessor in the LMC performs the following functions:

- Condensing status responses from the remote power controllers; since the status is the same as commanded except when a fault occurs in either the controller or the load.
- Filtering data; since many of the faults and status errors are of a transient nature, they can be resolved locally and need not enter the system data stream.
- Recording built-in-test data; only transmitting as required for central integrated test and as needed for maintenance.
- Processing load management priorities for the individual loads according to the priority level established by the central processor.
- Processing control equations to the extent of the information locally available so as to off-load the power system processor.

The LMC concept also promotes subsystem development wherein the subsystem unit can tap into the ring main power distribution bus, tap into the data bus, and operate autonomously except for minimal coordinating communication.

The technologies required have been verified in current developments and applications and provide a broad base of components and techniques from which to assemble a highly effective AAES for future aircraft.

5.3.2 AAES Hardware Development. - Lockheed has developed an alternative method for providing automatic load management and other advanced control concepts while avoiding the complexities of a MUX system. Its methodology is such that it can be designated a hardwired-AAES and may be more suitable for smaller or commercial aircraft. Such a system uses miniature-gage "dedicated" control wiring, in which each remote power controller (SSPC or RCCB) is simply direct wired for individual control. Thus the system has a certain basic simplicity in that it is wired in a manner similar to the conventional electrical installation. As there are no interfacing electronics and complex multiplex system between the signal sources and the controllers. The basic reliability should be equivalent to a conventional circuit the dedicated wiring retains the fault insulation of the conventional system and its

simplicity reduces the need for BIT and self-diagnostics. In more recent years, the Lockheed hardwired-AAES was given the acronym MALMS: Manual/Automatic Load Management System.

Figure 68 is a photograph of an AAES functional mock-up which incorporates the following:

- A current-limiting/display control panel to eliminate control circuit breakers.
- A miniature-gage control wiring system with current limiting to eliminate the problems of fault currents and fire hazards in the wiring.
- An automatic load management system to prioritize loads and prevent overloading of generators/overtorquing of engines.
- A wire programming capability to permit wiring reconfigurations and equipment updates.
- Quick attach/detach modular instrument panel assemblies.
- A Boolean processor, such as the Control Logic Assembly used in the S-3 aircraft.
- Solid state power controllers rated at 28 V 10 amp.
- Advanced hybrid ac power controllers with solid state front ends to interface with low level logic control system.

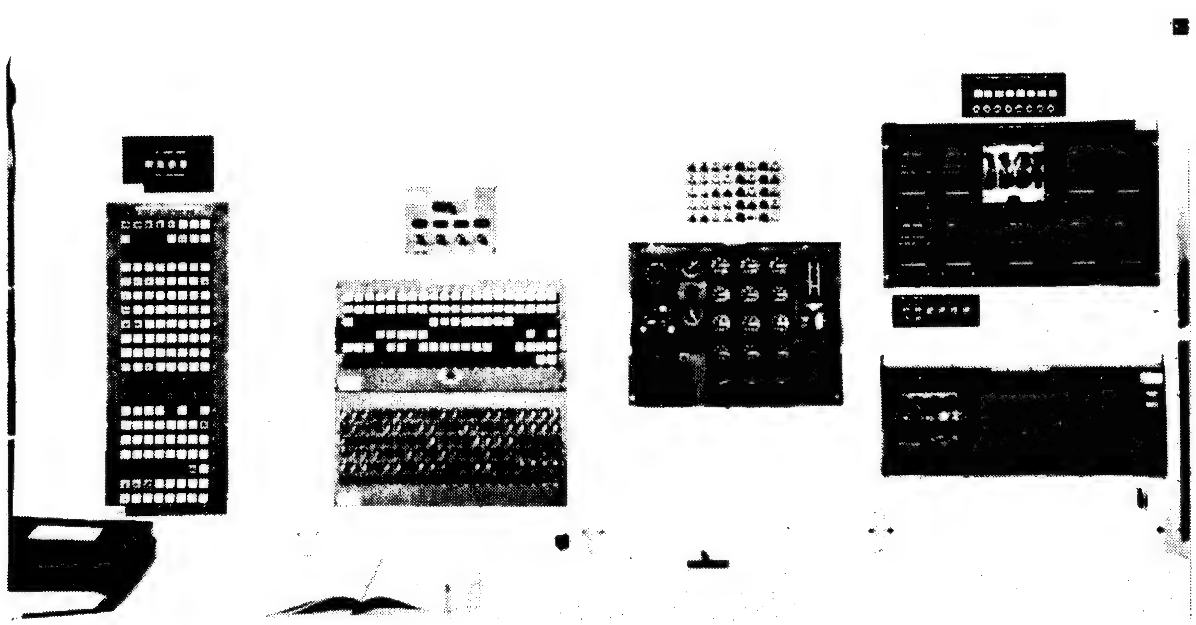


Figure 68. - Functional AAES mock-up.

- Power controller cabinet assembly to bring advanced thermal and management techniques to solid state devices.

Most of the above items/assemblies were reduced to pre-production format and the integrated system testing was performed validating the overall system efficacy of this Lockheed-AAES concept. The SSPCs were a significant item of development, but lack of funding prevented full scale qualification-type testing to be accomplished. The functional operation of the SSPCs under normal and abnormal loading conditions was established and verified during integrated system tests. These tests were performed using a miscellany of real and simulated loads. The real loads consisted of rotary/linear actuators, loads, heating dc motor ac fan motor and large power contactors.

The simplicity of the control circuit is illustrated in figure 69. In this schematic a typical control circuit breaker is replaced by a dual-lamped alternate-action push-switch. The lamp-switch permits tripping of the remote controllers in a fashion similar to pulling a control circuit breaker. The dual-lamp switch also doubles as a current limiting device while providing at the same time an ability to monitor the status of the remote power controller and flag its condition to the flight station. The final function of the lamp-switch is to provide a reset capability for the SSPC or RPC when it trips and latches out. Also, because of the low level currents required, the control circuit can utilize miniature gauge wire. This could be 26 gauge conventional wire, but there are many significant advantages to using very small gauge flat cables that offer mass termination techniques.

Since current limiting is effective in the control wiring, load management can be effected simply by grounding the control line of any RPC, so that it is deprived of a driving voltage. If many loads need to be disconnected this is accomplished simply by collecting the appropriate loadcontrol wires via isolation diodes to a bus which is grounded through power transistors. In the functional mock-up a quad power-transistor array was used to control four groups of loads, prioritized at different levels. Each of the four power transistors was controlled by a gating voltage developed by the load management logic assembly (LMLA). The LMLA itself as designed and fabricated by Lockheed, is shown in figure 70. This particular assembly which uses modular cards supervises and controls the operation of some 200 loads. The LMLA can however be extended to any number, simply by the addition of extra cards.

Selection of the priority level of individual loads can be effected by a joining of the LMLA with the programmable wiring module (PWM) as shown in figure 71. This module is an insertable/removable wire program that can handle a confluence of 360 wires. Again, the capacity is expandable by the addition of more Zero Insertion Force (ZIF) connectors. With the use of ZIF connectors, the wire wrapped program board can be inserted and removed with minimal force.

It is implicit in the hardwire control technology that it can be easily interfaced with remote ac or dc power controllers. The ac controllers, (which are modified electromagnetic contactors), are designed with a similar input impedance as the SSPCs so the ac and dc RPCs can be controlled by the same

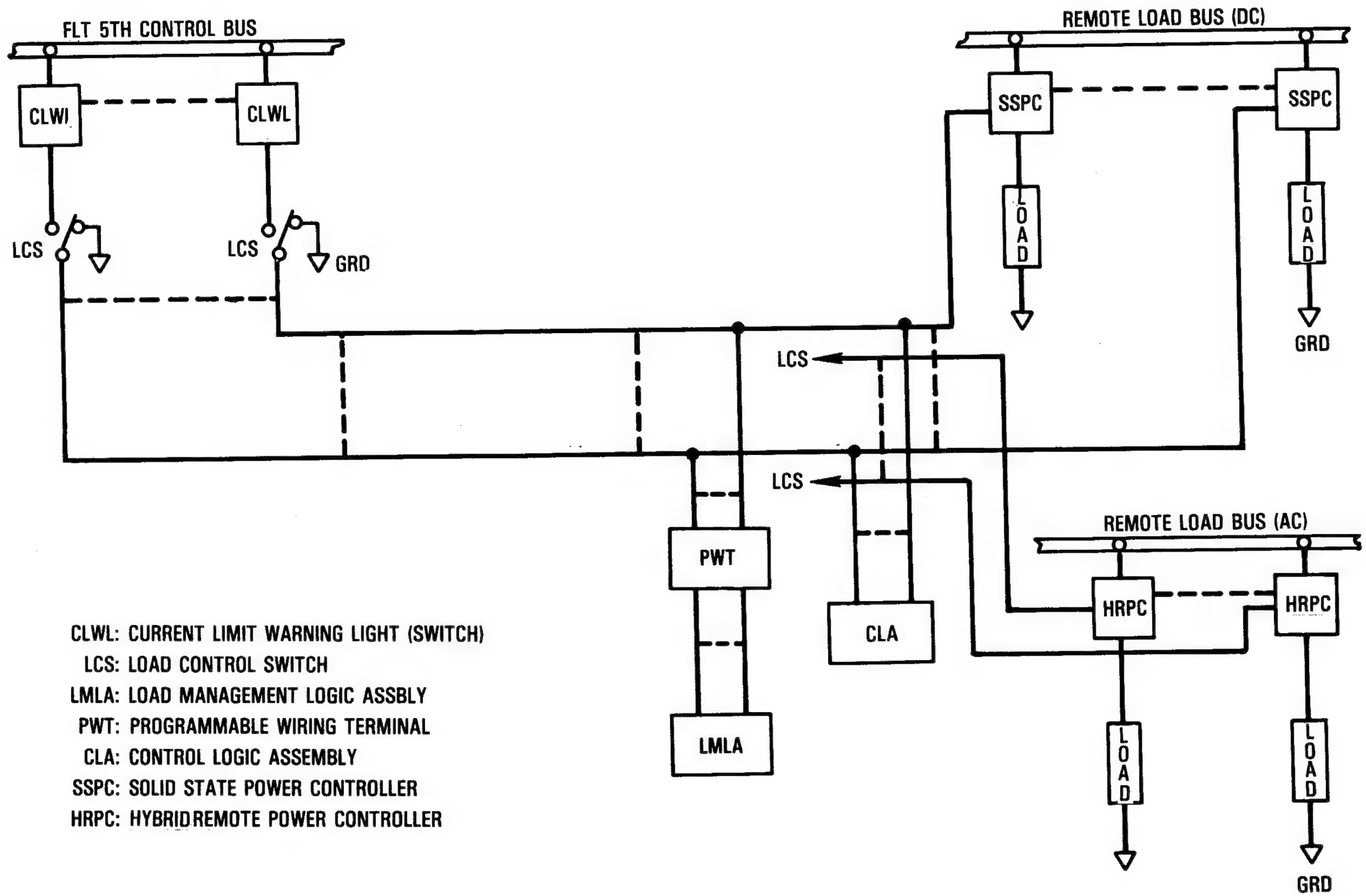


Figure 69. - Lockheed hardwired AAES system.

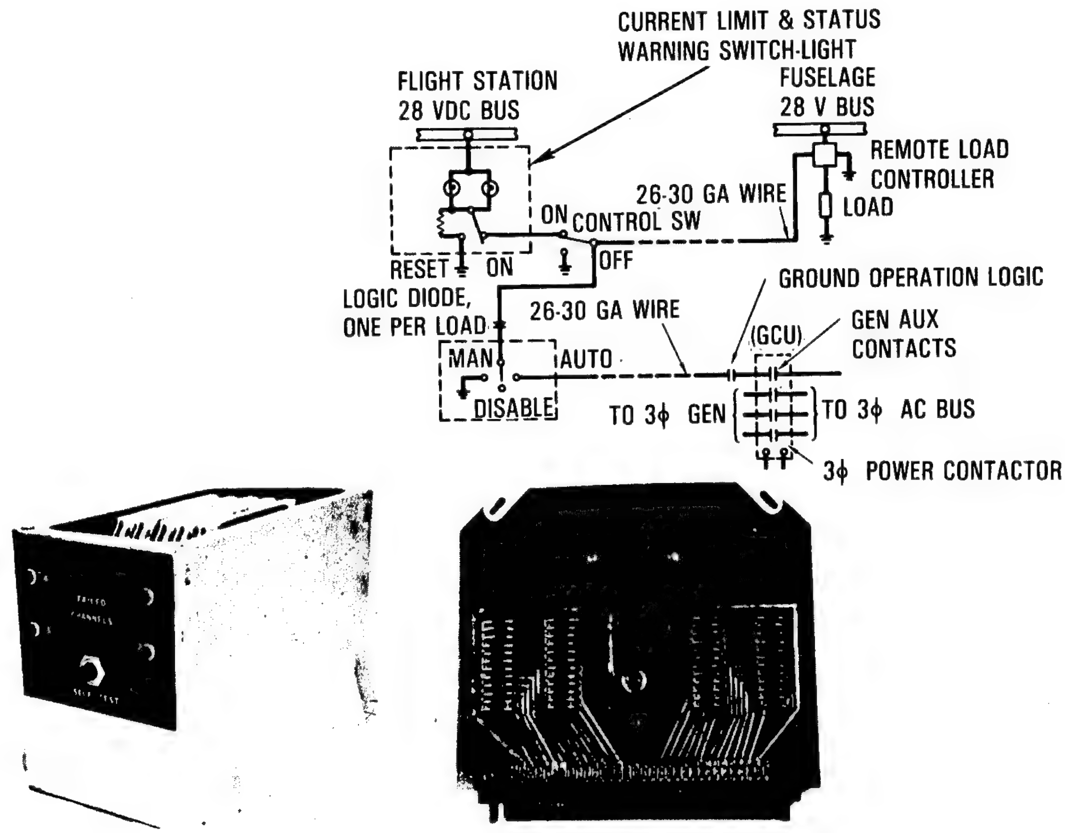


Figure 70. - Load management logic assembly.

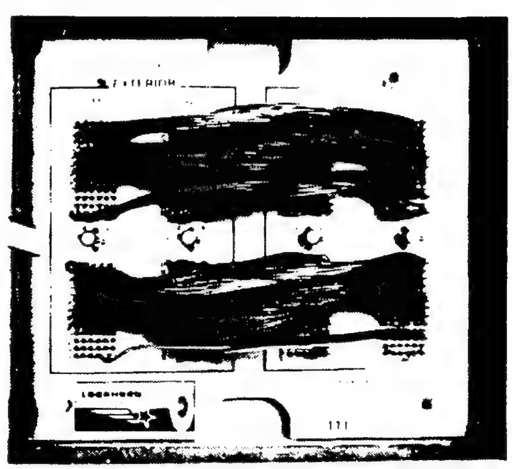


Figure 71. - Wire program board.

current of 1.0 milliamp or less. Since the controllers require a steady control current similar to a conventional relay, the controllers are less affected by transient voltages or any other electromagnetic anomaly. Such perturbations in a sequential digital control system could cause transient errors.

Thermal management and protection against voltage spikes were important design criteria for the SSPC cabinet assembly. This particular assembly provides a thermally and electrically benign environment for the SSPCs. Figure 72 is a photo of the SSPC cabinet assembly which accommodates six metal cored circuit boards, each of which carries eight 10 amp SSPCs. The novelty of the design is that the metal cored circuit board doubles as a heat conduction plane and a printed circuit wired module. These boards can be inserted/removed from the cabinet assembly without any electrical wire or mechanical screw/stud disconnections.

Voltage transient/spike-voltage protection for the SSPCs is provided at the mother board level by the use of two solid state surge-arrestors and two in-line chokes. The design objective of the spike voltage protection circuit was to select arrestors having an extremely fast "turn-on" time so that the SSPCs would not be exposed to any high voltage transient before the arrestors clamped the voltage to ground.

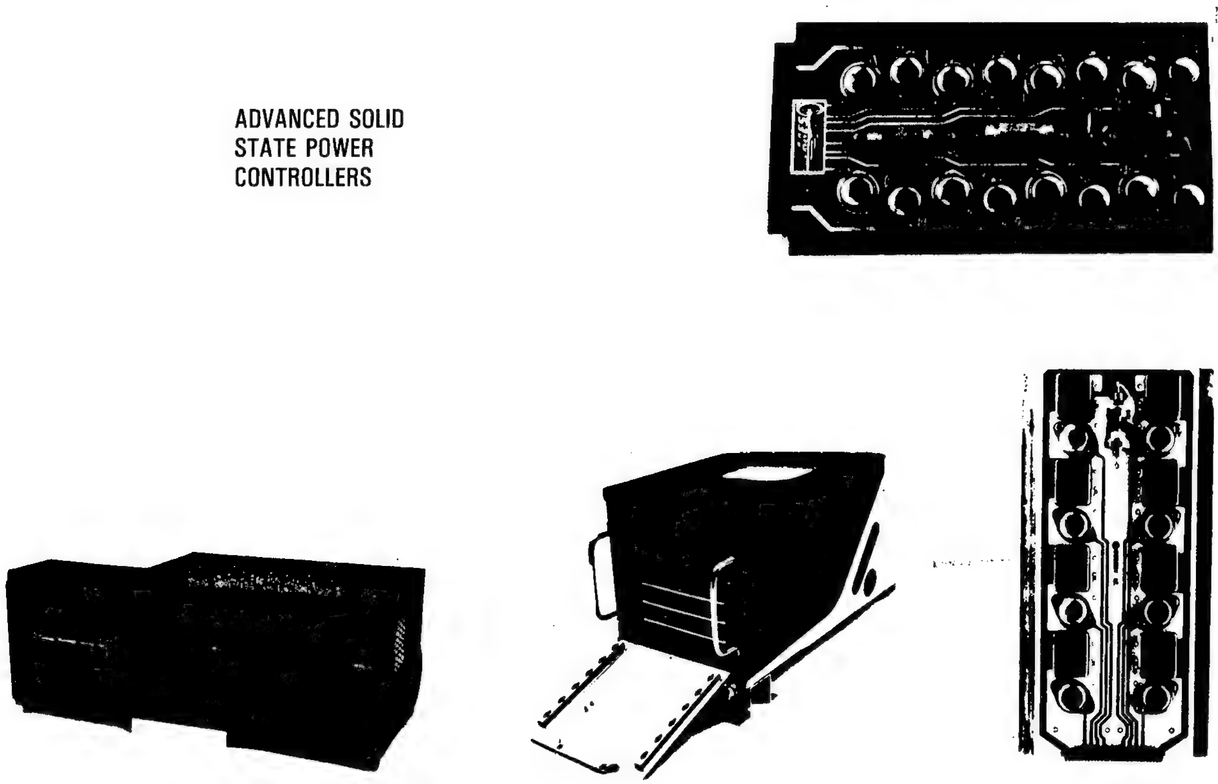


Figure 72. - SSPC cabinet assembly and SSPC circuit boards.

Figure 73 is a photo of the ac load controller (RPC) cabinet assembly, which mounts nine modules each consisting of three single-phase, 200 Vac, 10 amp relays and one three-phase, 200 Vac, 10 amp relay. These modules are quick-attach/detach assemblies that can be installed and removed from the front of the panel without disconnecting any wiring. The ac power controllers utilize solid state front end circuitry and auto reset thermal circuit breakers. As an additional refinement, protection against open-phase failure is also provided for the three-phase relays.

5.4 Digital Electric Management

The foregoing sections have described the technology and developments in advanced aircraft electric systems that can be applied to the IDEA. Because they are based on past and current activities, they have only addressed the requirements of conventional power systems and utilization equipment. The IDEA concept greatly expands the use of electric power, replacing hydraulic and bleed air power. These are primarily high power loads which strongly affect power generation and primary power busing which are covered in other portions of this study. But since there are relatively few of these additions, their effect on the Electric Power Management System is not a governing factor. Also, they are mostly autonomous subsystems so that the added load on the data bus and processor is not a problem. The following design and analysis can therefore directly apply to those technologies and developments of the previous sections.

As previously discussed, the broad AAES studies by NADC and AFWAL include generators, generator controls, power bussing, power conversion and special utilization equipments. These hardware and equipment advances, included in

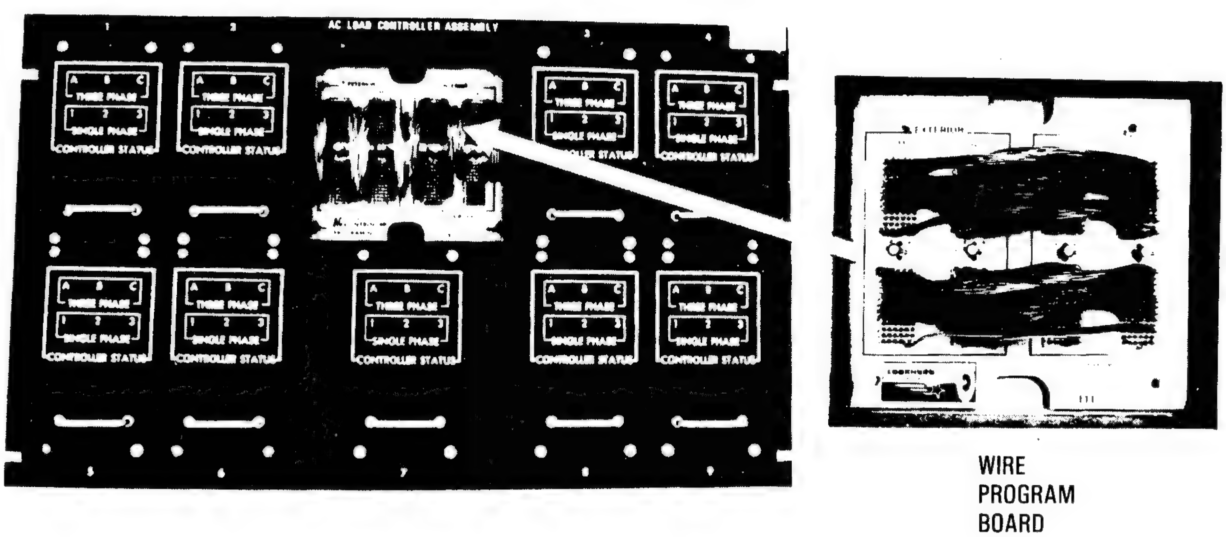


Figure 73. - Ac load controller assembly.

the IDEA, are beyond the scope of this section of the study with the possible exception of the generator controls. However, in all of the previous AAES studies it was found that the generator control functions and requirements were not compatible with EMUX systems. This has been generally resolved by treating the generator and its GCU as an autonomous subsystem with a remote terminal interface to the EMUX. The data handled by this GCU-RT would include generator status information required for power and load management and the control and status information for the integrated control/display units. With that approach then the AAES acronym is too broad to be used in the following system description. The EMUX acronym is also unsatisfactory because multiplexing is only one component of the system. A more descriptive acronym is Digital Power/Load Management System (DP/LMS).

Digital power control provides the basic advantage to advanced systems. The technology for the remote power controller is readily available and would be "off-the-shelf" if the quantity of applications justified it. In spite of the lack of application, the development programs have kept the technology up-to-date; improving characteristics, configuration, and cost effectiveness. The selection of solid state (SSPC) or hybrid (RPC) depends on their cost effectiveness so that the quantity of each is primarily based on economics at the time of commitment.

Digital data multiplexing (MUX) offers further advantages, but the change from discrete wiring to the complexities of a MUX system requires a major commitment and the attendant operations confidence. The hardwired Manual/Automatic Load Management System (MALMS) offers a method for attaining digital power control and load management without the complexities and expense of a MUX system. The simplicity of the MALMS approach could be an advantage for small commercial aircraft, particularly when MUX is not employed in any other systems. However, for a large aircraft such as investigated in this study, MUX is already justified in the avionics system and in the flight control system. Therefore a MUX system is recommended, particularly a MUX data bus per MIL-STD-1553 in accordance with the previous technical analysis.

The load management center (LMC) concepts provide an integration of the SSPCs with the MUX. The "smart" LMC is a natural outgrowth of this integration. Particularly since a microprocessor is generally used to implement the MUX interface protocol and data management for the subsystem interface. Therefore a "smart" LMC having a 16 bit microprocessor with adequate memory and software to do the local processing previously itemized is recommended for this IDEA Load Management System.

The power system processor (PSP) is the central component of any EMS, required to exploit the advantages of digital power control. The only trade-off involved is in the selection of the actual hardware and software at the time of implementation. Previous studies and implementations of EMUX systems have not shown a requirement for any special characteristics for power system processing.

The selection of the architecture of an DP/LMS for the IDEA is straightforward. The size of the aircraft and the quantity of sensors and loads justifies a split system approach. That is two partially independent EMUX

systems with a cross tie to coordinate the load/power management functions and to transfer data between the systems. Each PSP is the active primary processor and bus controller for its segment and acts as a condition monitor for the other segment. Each PSP also has standby capacity adequate to control the entire system in a fail-safe mode.

5.4.1 IDEA DP/LMS design. - The remote power controller selections for the Electric Management System of this IDEA study are listed in table 11. MIL-P-81653 is the basic specification for these power controllers. The majority of the controllers are solid state and the preferred configuration for commercial aircraft application is shown in figure 60. This multi-unit card is compatible with ARINC 600 box specifications. Figure 74 shows the circuit density that can be achieved, with up to 8 SSPCs per card. With this arrangement all required SSPCs, including spares, could be contained on 200 cards with an average of 16 multiunit SSPC cards per LMC. The electro-mechanical Hybrid Power Controllers and the bus contactors would be external to the LMC, except for the power source selector relays required for the internal SSPC power buses.

The load management center (LMC) would also contain the redundant RT to interface with the multiplex data bus. The intelligent LMC concept, as previously described, should be incorporated through redundant imbedded microprocessors. The microprocessors should be per MIL-STD-1750 if they become commonly available as planned. Full advantage of the local distributed processing should be taken, including normal and fault data filtering, BIT with nonvolatile memory, and partial control equation solution. It is estimated that 12 LMCs would be required to handle the 1000 typical loads and the added power control for the EMAs and FCS that were incorporated in this IDEA. In addition to power control, power conversion equipment should be included in the LMCs or in close proximity. By having local power conversion, the need for separate ac, dc, and other types of power bussing is eliminated. This is particularly effective since prioritized power bussing requirements are provided by the power management function. Also, since the controls and logic are electrically isolated from the power system, there is no longer any need for a centralized point of regulation (POR). Each LMC and the equipment it serves can be autonomous with its own POR. Finally, with local power conversion and the power management function, only a single power source is required - the redundant ring main bus.

Separate remote terminals (RT) are also required to pick up some of the scattered sensor signals. For sensors that are in the same area as the loads that they effect it is best to connect these sensors to the LMC, allowing local processing of those control equations. It is estimated that six RTs will be required for the EMS. Additional RTs will also be included in the EMS to exploit the multiplexing capabilities and equipments as a utility service for other subsystems. Multiplex data buses per MIL-STD-1553 are selected for the reasons given in the preceding sections.

Two power system processors (PSP) are required for the DP/LMS. There are no extraordinary requirements for these processors except that they incorporate two data bus control units (BCU) providing for primary control of one redundant data bus, and backup control of another. A 16 bit micro-processor

TABLE 11. - POWER CONTROLLER SELECTION

(BASED ON STANDARD AIRCRAFT LOADS)

Type	Min. Wire	Rating	Quantity
AC-SSPC	#26	1.0A	344
	#26	2.0A	74
	#24	3.0A	73
	#22	5.0A	46
	#20	7.5A	20
AC-HYBRID	#12	20A	4
AC-3ϕ	#20	7.5A	10
	#18	10A	10
	#16	15A	6
	#12	20A	6
DC-SSPC	#26	2.0A	296
	#24	3.0A	48
	#24	5.0A	25
	#20	7.5A	20
	#18	10A	14
	#16	15A	6
	#12	20A	4
DC-HYBRID	# 8	35A	14

per MIL-STD-1750 would provide adequate processing capability and should be employed if they become widely available as expected. The software for the PSP may also become standardized as far as the real time processing executive, such as the DAIS Executive. The actual control algorithms will have to be customized to the requirements of the IDEA. However, the control software should incorporate the new techniques described in the AAES Technology section.

The architecture of this DP/LMS (figure 75) provides two separate, redundant data buses. One reason for employing two buses is to accommodate the large capacity of the IDEA DP/LMS, including wiring into the wings and into the empennage. An objective is to limit the length of each to less than 300 feet. This also splits the data bus and processing loads while gaining extra redundancy in bus control and in processing (fail-safe mode). The LMCs and the RTs are distributed throughout the aircraft so that the sensor and load wiring is minimized. An average length of 20 feet for each wire segment is estimated for this system. Each data bus has capacity for 31 RTs although only 9 each are required for the electric system. Additional RT taps are shown in figure 75 to indicate RTs provided for the utility service.

Additional considerations in selecting the DP/LMS architecture and the DP/LMS itself include the added power and digital data distribution required

115V AC 1 PHASE CIRCUIT ARRANGEMENT				
CURRENT RATING	1 AMP PER UNIT	2 AMP PER UNIT	3 AMP PER UNIT	5/7. 5A. PER UNIT
UNITS PER CARD	8	6	4	2
1 – 2	PWR IN	PWR IN	PWR IN	PWR IN
3 – 4	OUT 1	OUT 1	OUT 1	OUT 1A*
5 – 6	OUT 2	OUT 2	NC	OUT 1A
7 – 8	OUT 3	OUT 3	OUT 2	OUT 1B
9 – 10	OUT 4	NC	NC	OUT 1B
11 – 12	NEUTRAL	NEUTRAL	NEUTRAL	NEUTRAL
13 – 14	NC	NC	NC	NC
15 – 16	SIG. GND.	SIG. GND.	SIG. GND.	SIG. GND.
17	CNTL 1	CNTL 1	CNTL 1	CNTL 1
19	CNTL 2	CNTL 2	CNTL 2	NC
21	CNTL 3	CNTL 3	NC	NC
23	CNTL 4	NC	NC	NC
25	15V DC	15V DC	15V DC	15V DC
27	15V DC	15V DC	15V DC	15V DC
29	CNTL 5	NC	NC	NC
31	CNTL 6	CNTL 4	NC	NC
33	CNTL 7	CNTL 5	CNTL 3	NC
35	CNTL 8	CNTL 6	CNTL 4	CNTL 2
37 – 38	SIG. GND.	SIG. GND.	SIG. GND.	SIG. GND.
39 – 40	NC	NC	NC	NC
41 – 42	NEUTRAL	NEUTRAL	NEUTRAL	NEUTRAL
43 – 44	OUT 5	NC	NC	OUT 2A
45 – 46	OUT 6	OUT 4	OUT 3	OUT 2A
47 – 48	OUT 7	OUT 5	NC	OUT 2B
49 – 50	OUT 8	OUT 6	OUT 4B	OUT 2B
51 – 52	PWR IN	PWR IN	PWR IN	PWR IN
TERMINAL	FUNCTION			

NOTE: EVEN NOS. 18 – 36 ARE ALL SIG. GND.
*NOTE: A = 5A., B = 7.5A

Figure 74. - SSPC circuit arrangement.

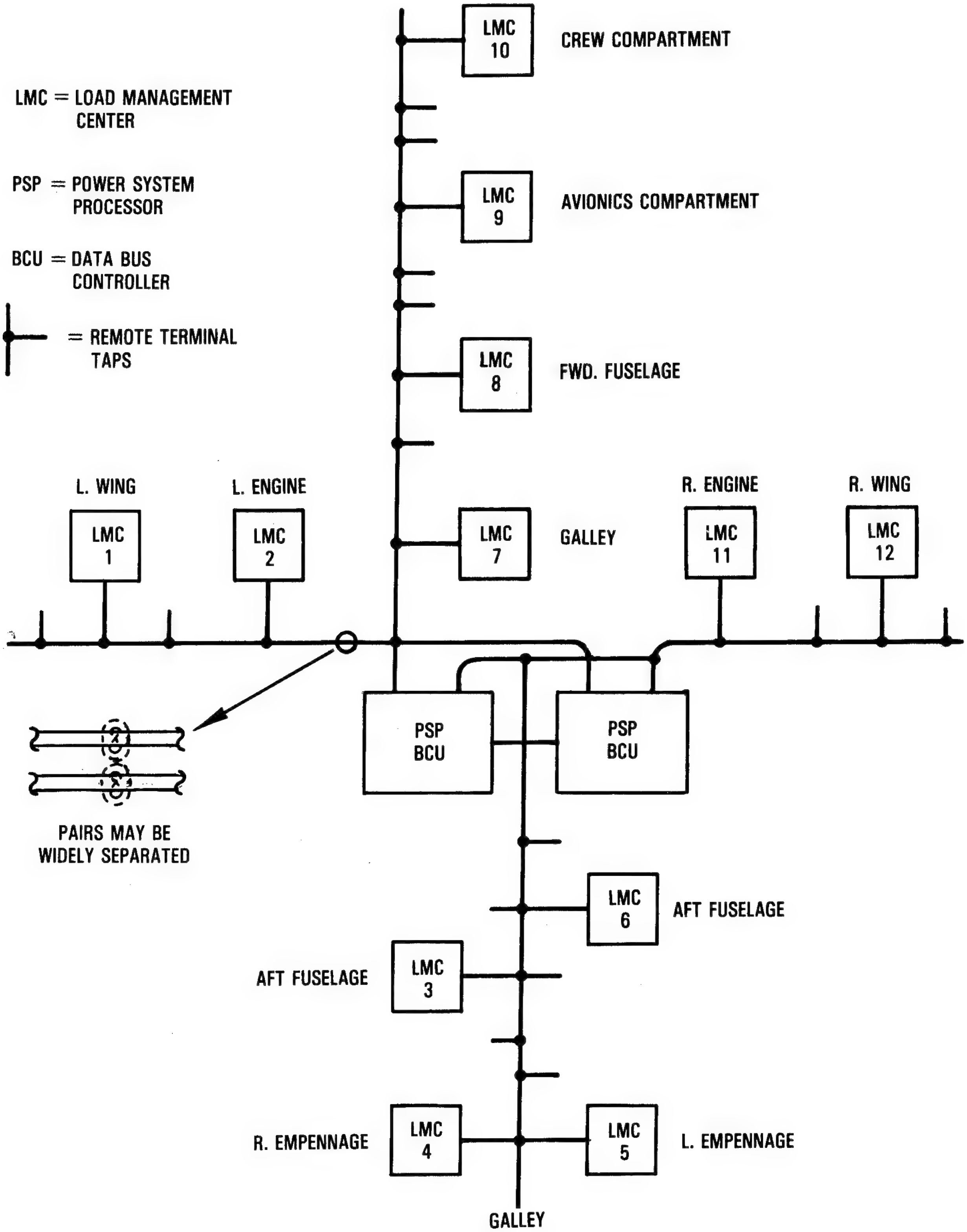


Figure 75. - DP/LMS architecture.

for the electric flight control system and the environmental control system. A method for providing highly reliable, redundant electrical power for the electromechanical actuator systems (EMAS) is shown in figure 76. Each of the LMCs can select power from either side of the wing loop bus tie breakers (BTB). Thus four isolatable power paths are available to each EMA through the two LMCs. An additional standby power source could also be provided for flight critical EMAS. This standby power source would not require any load management control so that the power connection would be made through a bus tap fuse module.

The DP/LMS can also serve as a standby or auxiliary digital data path for EMAS control as shown in figure 77. Here the primary EMAS control is provided by an independent flight control multiplex bus and RT, which may be additionally redundant. This would provide a triple redundant data path for a flight critical actuator.

5.4.2 DP/LMS weight analysis. - Weight reductions using the DP/LMS occur in several weight groups such as electrical system and flight controls. These benefits are therefore discussed in Section 12.2.3.

5.4.3 DP/LMS reliability analysis. - Reliability is a critical trade parameter but reliability models need to be based on the details of a specific system. For systems making extensive use of microelectronics MIL-HDBK-217C, RADC-TR-70-232, and RADC-TR-67-018 would be employed. High reliability in an EMUX system is attained through redudancy in the PSP, MUX, and RT/LMC as previously discussed. Details of the B-1 EMUX are available and the Harris Corporation analysis predicts an exceedingly good system reliability of 0.999992 for the typical B-1 mission. The mean time between corrective actions was assessed at 42 to 50 hours for the 36 boxes of the B-1 EMUX system.

Although the details of this proposed Digital Power/Load Management System are not complete, a good reliability estimate can be obtained by employing the analyzed values of the B-1 and the completed developmental systems. A reliability estimate of a conventional electric system consisting of individual control/power circuits may be obtained by assuming a linear relationship between the individual circuit failure rates and the number of circuits in the system. This same approach can be applied to the DP/LMS if the failure rates of common or shared hardware, such as the PSP, MUX, and RT/LMC, are accounted for in the power generation and power bussing analysis. The linear estimate also assumes that an "average" control/power circuit can be defined to determine a representative failure rate per circuit. Evaluation of previous detailed power system studies provides that an average conventional circuit consists of a control circuit breaker, two control switches or relay contacts, a power circuit breaker, and a power relay. The failure rate of these components plus the attendant wiring and connections sum to a circuit failure rate of 23.3 per million hours. With digital power control only one device, the RPC, is interposed between the power and the load. Also, the signal sources are passive, receiving power from the RT interrogation pulses, and the control logic is incorporated in the common hardware. The resulting failure rate of the digital power control circuit is only 5.2 per million hours. With

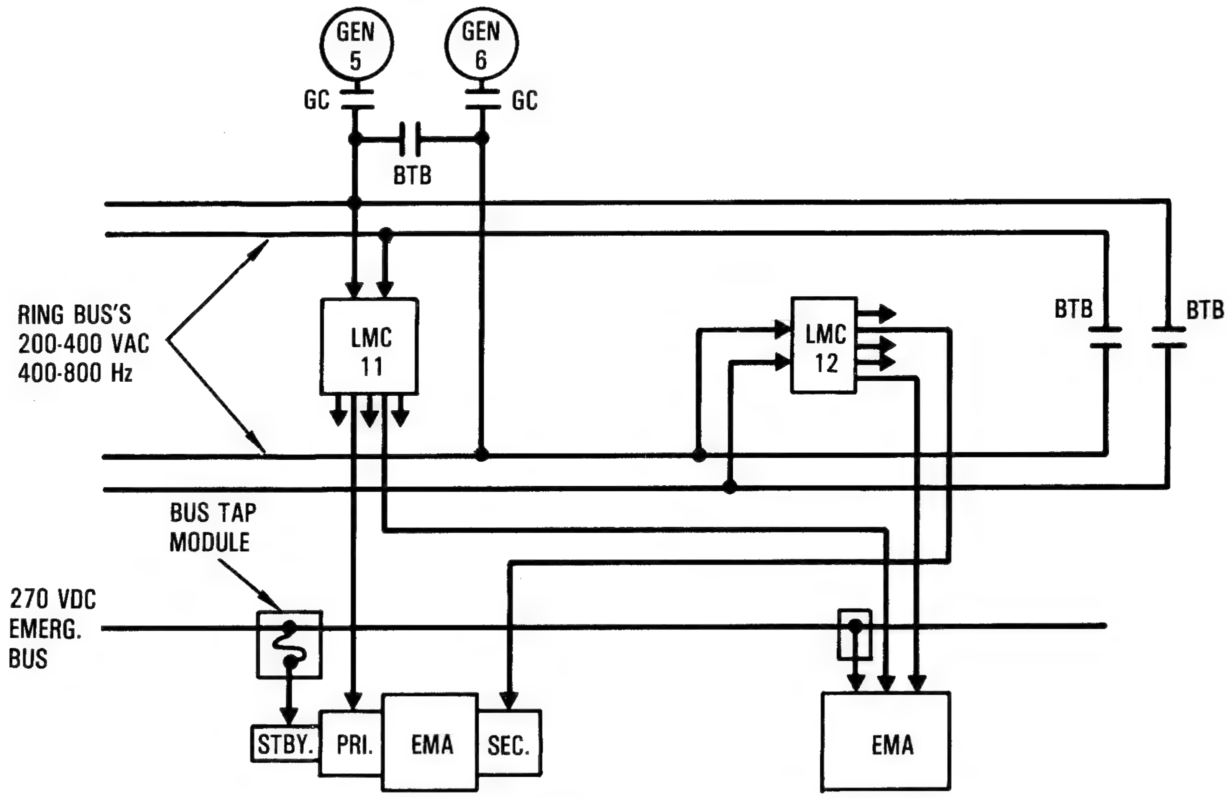


Figure 76. - EMAS power distribution.

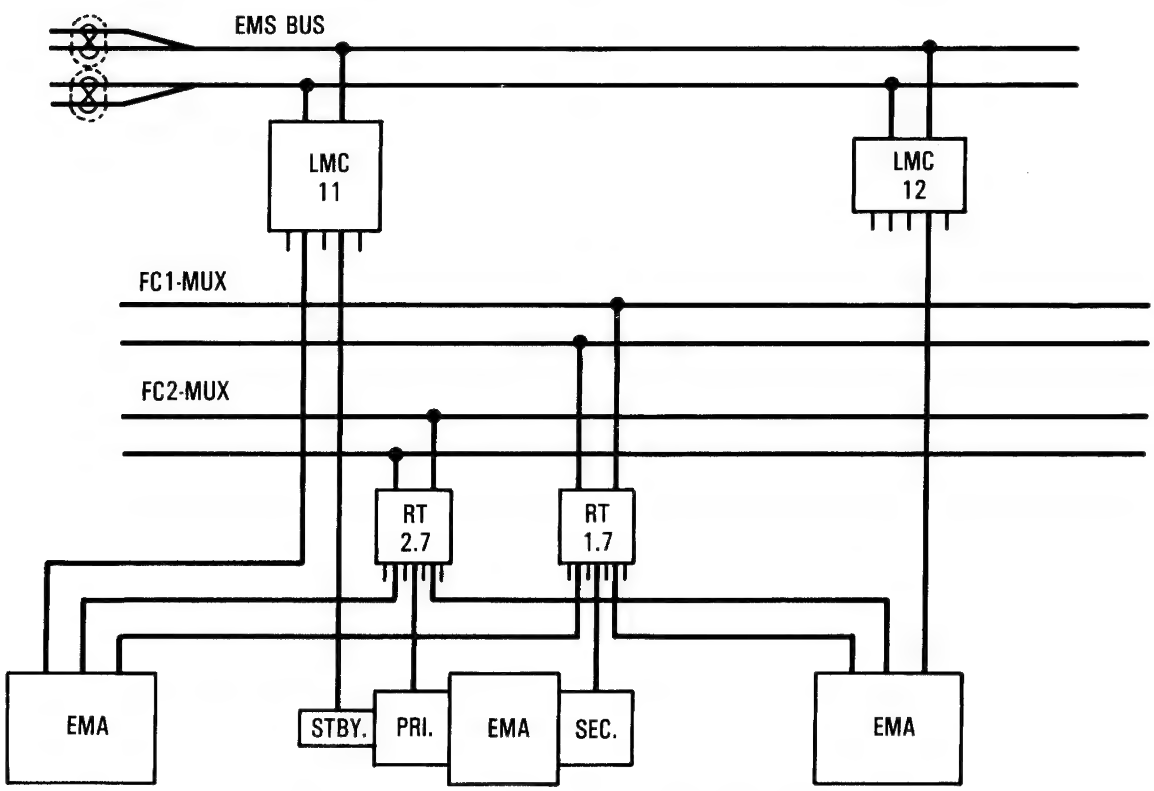


Figure 77. - EMAS data distribution.

more than 1000 circuits required, this improvement in circuit failure rate more than offsets the added common hardware in the DP/LMS.

The reliability estimates for the common hardware, to be added to the power generation and bussing analysis, are also required to estimate the life cycle cost of the DP/LMS. The power system processors as described are estimated to have an individual MTBF of approximately 5000 hours, but with fault tolerant design the effective failure rate is reduced to 50 per million hours. With two processors cross coupled, for fail-safe mode the probability of failure during a ten hour flight reduces the $(10 \times 50/10^6)^2$ or 0.00000025. The remote terminals, including those incorporated in the load management centers, are estimated to have an individual MTBF of approximately 10,0000 hours. But with dual redundancy, the effective failure rate is reduced to 5 per million hours.

Overall, the increased reliability of the DP/LMS approach can be attributed to the reduction of component units by the sharing of highly reliable microelectronic circuitry and the low failure rate of solid state power controllers.

5.4.4 DP/LMS cost analysis. - The DP/LMS features an incentive for its application and many of its advantages also provide a direct cost savings. In first order is the weight savings, shown in the previous section, which reduce the direct operating cost (DOC). The DP/LMS approach reduces not only weight of wiring but also quantity, thus reducing harness costs. Another advantage of the DP/LMS approach is clearly related to the increased reliability, shown in the previous section. This is reflected in lower maintenance and replacement costs as well as a decrease in cancelled flights.

High labor costs also favor the adoption of DP/LMS for commercial aircraft, not only due to the improved reliability but also the reduction in time for maintenance. This results from the extensive built-in-test (BIT) provided by DP/LMS. The BIT along with the modularity of the design concepts significantly reduces the man hours per maintenance action as compared to a conventional system.

Closer examination of the composite costs results in the following observations;

(a) The greatest cost of the EMS is the relatively well defined acquisition expense.

(b) The substantial projected savings from reduction of maintenance and cancelled flights also indicates a higher availability and further value or lower LCC for the EMS.

5.5 Plans and Resources

The total concept of digital data magenement, digital power control, and digital load management, interfaced with solid state/hybrid power controllers, is in a very advanced stage of development. Many suppliers are involved with

the design, development and manufacture of the component elements, while others are involved with the systems technology and the integration of specific parts of the total systems technology. Another type of systems management and integration is that undertaken by Vought-LTV, who working under NADC/AFWAL contracts, was responsible for integrating, in a total sense, a complete digital electric system in mock-ups of an A7 airplane. A similar hot bench mock-up of a digital power/load management system for an advanced technology aircraft is imminently due for demonstration at NADC Warminster to add to the previous system integration experience on the A7 mock-ups.

Relative to the foregoing, NASA resources are not required on a major scale to ensure the maturation of the technology, but there are incipient systems integration problems that warrant NASA's continuing interest. This is particularly the case in the area of solid state and hybrid power controllers. There is also the question as to whether the cost, complexity, and sophistication of the technology in the advanced military weapon systems is warranted for the commercial transport and, particularly, for the small commuter type transport. Commercial airlines would welcome the benefits of digital control and advanced power management techniques, but they cannot afford the system design philosophy, espoused for the military airplane. Therefore, NASA should consider the sponsorship of simpler alternative systems that would not involve a MIL-STD-1553B data bus and the unequivocal commitment of the technology to all solid state control (via contactless switches, solid state, and hybrid power controllers). Since the existing technology has already been substantially supported by military funding, it appears essential that NASA should consider the support of an alternative system that would be more dedicated to the specific needs and requirements of commercial aircraft.

6. ACTUATION SYSTEMS FOR THE IDEA AIRPLANE

6.1 Flight Control Actuation Options

The flight control actuation functions were evaluated first since it was felt that these functions represented the most rigorous applications which occurred on the airplane and, thus, would act as a "bellwether" and guide for all the other aircraft actuation functions. Each potential actuation approach was evaluated against the static and dynamic performance requirements listed in table 12. Forty-four types of systems falling in eleven categories were considered and are described in table 13. It can be seen from this table that the diverse elements making up the various systems were highly eclectic and ranged from torque summed electric motors through infinitely variable traction transmissions to integrated actuator packages. However, each of the eleven categories was characterized by certain specific functional elements arranged in a specific sequence as follows:

Category Characteristic

1. Power Inverter - Bidirectional Electric Motor - Power Hinge

2. Power Inverter - Bidirectional Electric Motor - Screwjack

TABLE 12. - EMAS DESIGN/PERFORMANCE DATA

FLIGHT SURFACE	SPOILERS						AILERONS		RUDDER	STABILIZER/ ELEVATOR
	1	2	3	4	5	6	I/B	O/B		
Loads: ft. lb.										
• Stall	9.8K	4.7K	4.2K	2.7K	3.2K	1.9K	5.9K	7.3K	9.7K	80.2K
• Design	6.5K	2.5K	2.2K	1.42K	1.5K	0.86K	4.0K	4.7K	7.2K	20.0K
Rate: Deg/Sec	60	60	60	60	60	60	35	35	30	7.3
Displacement (Max)										
• Degrees	-60	-60	-60	-60	-60	-60	±20	±20	±30	+1 -14
• Inches	5.59	5.59	3.05	3.05	2.65	2.65	±2.48	±1.46 Ret ±1.38 Ext	S 25.3 L 30.0	38.62
Freq. Response										
• Bandwith (-3dB)	3.2	3.2	3.2	3.2	3.2	3.2	3.2	3.2	1.6	2.2
• Amplitude (Deg)	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.0	0.3
Min. Servo Stiff (in.lb/Deg)	6.3K	6.3K	1.75K	1.99K	1.23K	1.68K	3.49K	8.7K	60K	2.61×10^6
Inertia (in.lb.sec^2) (Surface)	89.8	69.4	10.2	19.7	23.1	17.8	3.18K	1.28K	0.629K	41.51K
No. of Actuators	1	1	1	1	1	1	3	2	3	4
Resolution (Deg.)	0.25	0.25	0.25	0.25	0.25	0.25	0.1	0.1	0.1	0.01

NOTES

1. Except as noted otherwise, values shown relate to single actuators.
2. Spoilers have different amplitudes during different operational modes.
3. Rudder hinge moment limited to 5000fp at 260 kCAS: 8° Max./ flaps up.
4. Mechanical shear-section required at 100 kfp in stabilizer actuator outputs.
5. Values shown, per actuator, are maximum (under any failure mode).
6. Linear travel relates to hydraulic actuators: same as L-1011-Basic except for stabilizer/elevator.
7. Rudder deflection > 3.5° at 350 KCAS.

TABLE 13. - CANDIDATE EMAS SYSTEMS (PAGE 1 OF 2)

		Input Power Control		Input Power Transducer		Failure Protection Devices		Power Distribution Devices		Power Output Device	
Category	Sys. Type	Item	Req.	Item	Req.	Item	Req.	Item	Req.	Item	Req.
	1	Pwr Inv	1	BDEM	1	—	—	RGB, SHFT	1	MPH	1
	2	Pwr Inv	2	BDEM	2	BRK	2	DRGB, SHFT	1	MPH	1
	3	Pwr Inv	2	BDEM	2	DOG CLT	2	RGB, SHFT	2	MPH	1
1	4	Pwr Inv	3	BDEM	3	BRK	3	DRGB, SHFT	1	MPH	1
	5	Pwr Inv	4	BDEM	4	BRK	4	DRGB, SHFT	2	MPH	1
	6	Pwr Inv	4	BDEM	4	DOG CLT/BRK	2/4	DRGB, SHFT	2	MPH	1
	7	Pwr Inv	1	BDEM	1	DOG CLT	1	RGB, SHFT	1	MPH	1
	8	Pwr Inv	1	BDEM	1	—	—	RGB, SHFT	1	SCJK	1
	9	Pwr Inv	2	BDEM	2	BRK	2	DRGB, SHFT	1	SCJK	1
	10	Pwr Inv	2	BDEM	2	DOG CLT	2	RGB, SHFT	2	SCJK	1
2	11	Pwr Inv	3	BDEM	3	BRK	3	DRGB, SHFT	1	SCJK	1
	12	Pwr Inv	4	BDEM	4	BRK	4	DRGB, SHFT	2	SCJK	1
	13	Pwr Inv	4	BDEM	4	DOG CLT/BRK	2/4	DRGB, SHFT	2	SCJK	1
	14	Pwr Inv	1	BDEM	1	DOG CLT	1	RGB, SHFT	1	SCJK	1
	15	Pwr Inv	2	TSEM (2IN1)	1	—	—	RGB, SHFT	1	MPH	1
	16	Pwr Inv	3	TSEM (3IN1)	3	—	—	RGB, SHFT	1	MPH	1
3	17	Pwr Inv	4	TSEM (2IN1)	2	—	—	RGB, SHFT	2	MPH	1
	18	Pwr Inv	4	TSEM (4IN1)	1	—	—	RGB, SHFT	1	MPH	1
	19	Pwr Inv	2	TSEM (2IN1)	1	—	—	RGB, SHFT	1	SCJK	1
	20	Pwr Inv	3	TSEM (3IN1)	1	—	—	RGB, SHFT	1	SCJK	1
4	21	Pwr Inv	4	TSEM (2IN1)	2	DOG CLT	2	RGB, SHFT	2	SCJK	1
	22	Pwr Inv	4	TSEM (4IN1)	1	—	—	RGB, SHFT	1	SCJK	1
	23	Pwr Inv	4	BDEM	1	—	—	RGB, SHFT	1	ECC	1
5	24	Pwr Inv	2	BDEM	2	BRK	2	DRGB, SHFT	1	ECC	1
	25	Pwr Inv	3	BDEM	3	BRK	3	DRGB, SHFT	1	ECC	1
	26	Pwr Inv	4	BDEM	4	BRK	4	DRGB, SHFT	2	ECC	1
	27	Pwr Inv	2	TSEM (2IN1)	1	—	—	RGB, SHFT	1	ECC	1
6	28	Pwr Inv	3	TSEM (3IN1)	1	—	—	RGB, SHFT	1	ECC	1
	29	Pwr Inv	4	TSEM (2IN1)	2	DOG CLT	2	RGB, SHFT	2	ECC	1
	30	TTS	2	UDEM	2	REC CLT	2	RGB, SHFT	2	MPH	1
7	31	TTS	4	UDEM	4	REC CLT/BRK	2/4	DRGB, SHFT	2	MPH	1
	32	TTS	2	UDEM	2	REC CLT	2	RGB, SHFT	2	SCJK	1
	33	TTS	4	UDEM	4	REC CLT/BRK	2/4	DRGB, SHFT	2	SCJK	1
8	34	TTS	1	UDEM	1	—	—	RGB, SHFT	1	SCJK	1
	35	TTS	1	UDEM	1	—	—	RGB, SHFT	1	MPH	1

TABLE 13. - CANDIDATE EMAS SYSTEMS (PAGE 2 OF 2)

Category	Sys. Type	Input Power Control		Input Power Transducer		Failure Protection Devices		Power Distribution Devices		Power Output Device	
		Item	Req.	Item	Req.	Item	Req.	Item	Req.	Item	Req.
	36	Pwr Inv	1	DV	1	—	—	SHFT	1	—	—
9	37	Pwr Inv	2	DV	2	DOG CLT	2	SHFT	2	—	—
	38	Pwr Inv	3	DV	3	DOG CLT	3	SHFT	3	—	—
10	39	Pwr Inv	2	BDEM	2	BRK	2	AJGA	1	PPL	2
	40	Pwr Inv	4	BDEM	4	BRK	4	AJGA	2	PPL	4
	41	Switch	1	UDEM	1	—	—	SIAP	1	—	—
11	42	Switch	2	UDEM	2	—	—	DIAP	1	—	—
	43	Switch	3	UDEM	3	—	—	TIAP	1	—	—
	44	Switch	4	UDEM	4	—	—	QIAP	1	—	—

DEFINITION OF ACRONYMS

AJGA	Anti Jam Gearbox Arrangement
BDEM	Bi Direction Electric Motor
BRK	Electrically Operated Brake - Spring Loaded On
DIAP	Duplex Integrated Actuator Package
DOG CLT	Dog Clutch - Disconnectable Under Load
DRGB	Differential Reduction Gearbox
DV	Dynavector Transmission - Electrical, High Torque Low Speed Output
MPH	Mechanical Power Hinge
PPL	Push Pull Link
PWR INV	Electrical Solid State Digital Power Inverter
QIAP	Quadruplex Integrated Actuator Package
REC CLT	Reconnectable Clutch - Reconnectable Under Load
RGB	Reduction Gearbox
SCJK	Screwjack - Ball or Transrol Type
SHFT	Power Transmission Shaft
SIAP	Simplex Integrated Actuator Package
TIAP	Triplex Integrated Actuator Package
TSEM (2IN1)	Torque Summing Bi Directional Elect. Motors - Two on Common Shaft
TSEM (3IN1)	Torque Summing Bi Directional Elect. Motors - Three on Common Shaft
TSEM (4IN1)	Torque Summing Bi Directional Elect. Motors - Four on Common Shaft
TTS	Traction Transmission Servo - Infinitely Variable Bi Directional Output

3. Power Inverter - Torque Summing Bidirectional Electric Motor - Power Hinge

4. Power Inverter - Torque Summing Bidirectional Electric Motor - Screwjack

5. Power Inverter - Bidirectional Electric Motor - Eccentuator

6. Power Inverter - Torque Summing Bidirectional Electric Motor - Eccentuator

7. Infinitely Variable Traction Transmission - Unidirectional Electric Motor - Power Hinge

8. Infinitely Variable Traction Transmission - Unidirectional Electric Motor - Screwjack

9. Power Inverter - Dynavector

10. Power Inverter - Bidirectional Electric Motor - Anti Jamming Gearbox Arrangement

11. Integrated Actuator Packages

A discussion of each of the more important elements making up the various systems follows.

6.1.1 Power inverter. - The power inverter used for most of these applications consists of a controller and a power handling device. The function of the controller is to accept input signals from the flight control system (i.e., from the pilot and/or the central air data system) and deliver a control signal to the power handling device (usually an inverter) which latter signal has been modified by feedback data from the inverter itself and from the actuator output. In effect the controller interfaces with the command input, the primary electrical power source, the actuator motor, and an output function feedback device. It provides low-level signal system communication and high power switching and electrical control. Thus, a typical controller would include a junction for summing input command and feedback signals, and might also include such additional features as motor current limiting and pulse width modulation (PWM) to control motor speed and torque as a function of the sensed error amplitude. The controller would employ all-digital techniques and would be based on the use of microprocessors. The power handling device (inverter) would deliver controlled power to the actuators. It generates a synthesized rotating field as a function of input/output command errors and contains commutation and current control switching logic, power switch driver electronics, power switches, and current limiting. The power switching devices may be either SCRs or power transistors.

For the purposes of the electromechanical actuation system (EMAS) evaluation it is assumed that a 270 Vdc pulse width modulated inverter based upon liquid cooled field effects transistors (FETs) is representative of the

type of power inverter which would be used for all applications calling for this type of device.

6.1.2 Bidirectional electric motors. - Numerous different types of electric motors will be used on the IDEA aircraft. However, where bidirectional motors are concerned, the field of selection narrows considerably and, particularly where the application involves meeting flight control duty cycle types of requirements (i.e., servomotor requirements), the field narrows to essentially one type of motor. This is the rare earth-cobalt permanent magnet type motor based on the "inside out" type of design. Numerous studies (references 11, 12, and 13) have consistently indicated that this is the type of design to be preferred based on the following general set of advantages.

1. Up to five times the acceleration capability of the best wound rotor designs at equal power inputs.

2. At least twice the power rate of an equal frame size wound rotor design.

3. Relatively low commutation noise.

4. Because of its "inside out" design with all significant heat rejection occurring in the stator, the problems associated with heat rejection in the rare earth-cobalt permanent magnet motor are greatly reduced relative to other types of motor designs.

5. Superior stall-torque holding capability.

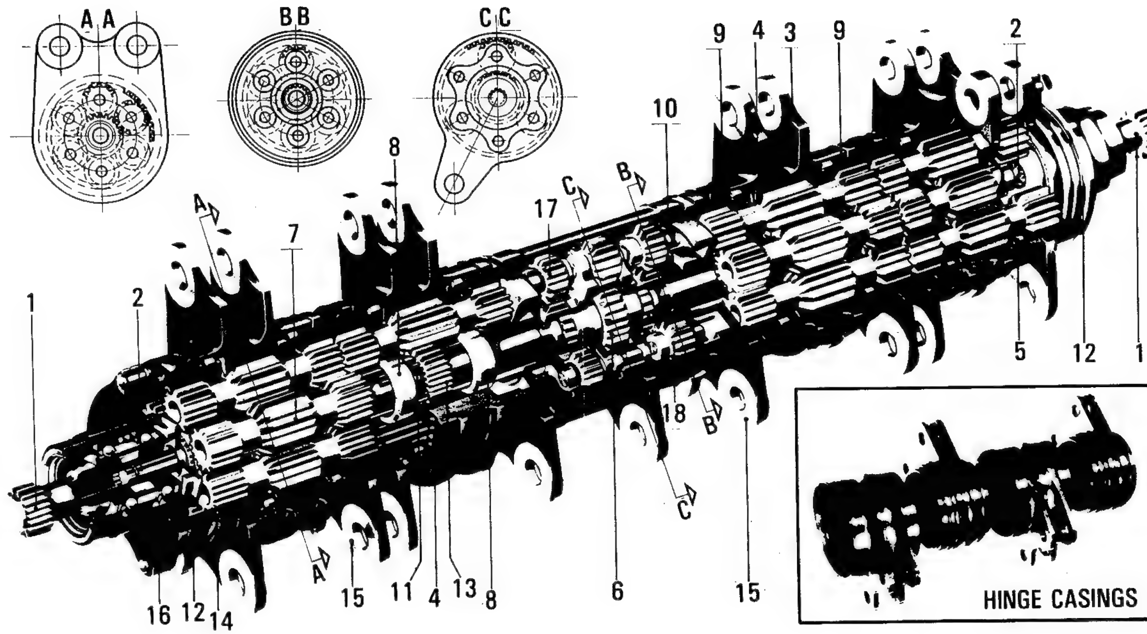
The frequency response requirements of the various flight control actuation functions used on large commercial aircraft will range from 3 to 12 Hz with subsequent designs crowding the upper end of this scale. Since power rate, as discussed in item 2, is the major determinant of a motor's ability to meet frequency response requirements, the fact that rare earth-cobalt permanent magnet motors have a 2 to 1 advantage is of extreme importance.

6.1.3 Hingeline actuators. - The hingeline actuator is one of two basic power devices that has been used to implement EMAS development (the screwjack is the second). As such, a considerable fund of development data has been generated to define its capabilities when operated as a part of an EMAS system. Figure 78 is a detailed drawing of the hingeline actuator showing it as a device comprised of a sun gear power input, multiple planetary spindle gears, and four annular ring gears. It is different from the conventional planetary gear arrangement in the following respects:

- The planetary spindle gears are free and independent - no carrier is used.

- Many spindle gears are used to distribute the load into the ring gear.

- Two reaction ring gears are positioned on either side of the output ring gear and thus the tangential tooth loads in the center mesh are reacted, in a balanced fashion, by the outer meshes.



FUNCTIONAL DESCRIPTION

THE DRIVE IS TRANSMITTED FROM THE INPUT SHAFT (1) BY A DRIVE SHAFT (2) TO THE FIRST STAGE SUN GEAR (6). IDENTICAL DRIVE SHAFT ARRANGEMENTS CAN BE PROVIDED AT BOTH ENDS OF THE TRANSMISSION TO PROVIDE A THROUGH DRIVE.

THE OUTPUT OF THE FIRST STAGE GEARBOX IS TRANSMITTED THROUGH TWO SECOND STAGE DRIVE SHAFTS (11) WHICH EXTEND TO THE OUTER SLICES OF THE ACTUATOR. EACH SHAFT (11) DRIVES THE SUN GEARS (13) OF THE IDENTICAL SLICES OF SECOND STAGE EPICYCLIC GEARS DISPOSED ALONG THE SHAFT.

EACH SECOND STAGE EPICYCLIC GEAR SET HAS SIX COMPOUND PLANET GEARS (7) SUPPORTED RADIALLY BY TWO SUPPORT RINGS (8). THE SUPPORT RINGS ALLOW THE PLANET GEARS SOME DEGREE OF ROTATIONAL FREEDOM TO TAKE UP AN OPTIMUM LOAD-SHARING DISPOSITION.

EACH COMPOUND PLANET GEAR IN THE SECOND STAGE SLICES HAS THREE TOOTH SECTIONS. THE OUTER SECTIONS ARE IDENTICAL AND MESH WITH ANNULAR RING GEARS (14) IN THE FIXED MEMBERS (15). THE CENTER TOOTH SECTION MESHES WITH ANNULAR RING GEARS (4) OF THE OUTPUT MEMBERS (3). A LARGE REDUCTION RATIO IS OBTAINED BY HAVING SIMILAR BUT DIFFERENT RATIOS OF PLANET TEETH TO ANNULUS TEETH FOR THE CENTRE AND OUTER SECTIONS. THE OVERALL REDUCTION RATIO OF THE TRANSMISSION IS TYPICALLY 800:1 WITH A FIRST STAGE REDUCTION OF ABOUT 10:1.

THE FIXED AND OUTPUT MEMBERS ARE HELD TOGETHER FOR TRANSPORTATION PURPOSES BY THE CENTRE DRIVE ROD ASSEMBLY, THE OUTPUT MEMBERS BEING SUPPORTED BY ROLLER BEARINGS (5). ROLLER BEARINGS ARE USED IN PREFERENCE TO BALL BEARINGS IN ORDER TO GIVE THE TRANSMISSION GREATER FLEXIBILITY IN BENDING. ROLLER BEARINGS ALSO CONFER GREATER AXIAL TOLERANCES FOR THE PURPOSES OF ATTACHMENT TO THE MOUNTING BRACKETS. STATIC (10) AND DYNAMIC (9) ENVIRONMENTAL SEALS ARE FITTED.

THE END CAP (16) IS SECURED IN THE FINAL SECOND STAGE FIXED MEMBER (15) BY A FLEXIBLE WIRE ROD (12) THREADED FROM THE EXTERIOR AND LOCKED IN POSITION.

Figure 78. - Typical hingeline actuator design.
(Courtesty Lucas-Aerospace)

- Two radial reaction plain rings are provided on each side of the central mesh to radially preload the spindle gear-ring gear meshes and to react the radial component of the gear mesh loads, as fed through the spindle gears, in a balanced manner. No planet support bearings are used.

- The "two-tooth-difference" between the "drive" and "driven" members, provides a very high harmonic drive type gear reduction.

- It has a very high effective torsional stiffness.

6.1.4 Screwjack. - The screwjack is the second of the two basic power transmission types considered for the EMAS system. The primary advantage of the screwjack is that it can act as a direct replacement for a hydraulic actuator. It is usually able to be installed in the same location and use the same end attach points as a hydraulic actuator. There are three basic types of ball screw devices, the screw thread type, the recirculating ball type, and the Transrol roller type. The screw thread type was eliminated from consideration for the purposes of this study because its efficiency is poor and it is only satisfactory for extremely low power functions in airplane applications. The roller type has exceptional load carrying capacity at the expense of some envelope penalties. The ball screw type is the most flexible and versatile and has by far the most development background behind it so it was used as the basic power interface where a linear actuator was indicated. Although ball screws can be designed with various degrees of end freedom, most airplane applications demand spherical bearings at both ends of the actuator. To meet this condition the so called "swallowed screw" design must be used exclusively. Therefore, "swallowed screw" designs similar to that shown in figure 79 were used for all ball screw applications in this study.

6.1.5 Torque summing bidirectional electric motor. - Torque summing can be accomplished in any one of at least three ways:

1. By using multiple motors geared to a common shaft.

2. By using multiple motors mounted linearly along the hingeline so as to have a common shaft.

3. By using multiple torque summing in which multiple windings in the stator, powered by different systems, drive a common multipole permanent magnet rotor.

Of the three approaches, approach #1 is the most versatile and it fits into various envelopes, but is the heaviest and the least efficient. Approach #2 is intermediate, having a long/inflexible envelope and a weight between approaches #1 and #3. Its power transmission efficiency is better than approach #1 and it is on a par with approach #3. Approach #3 is also inflexible in envelope and tends to be large in diameter although less than approaches #1 and #3 in total volume. Approach #3 is the lightest of the three approaches by 10 to 20 percent. Because of its "intermediate" status approach #2 was eliminated from serious consideration in subsequent trade study activities.

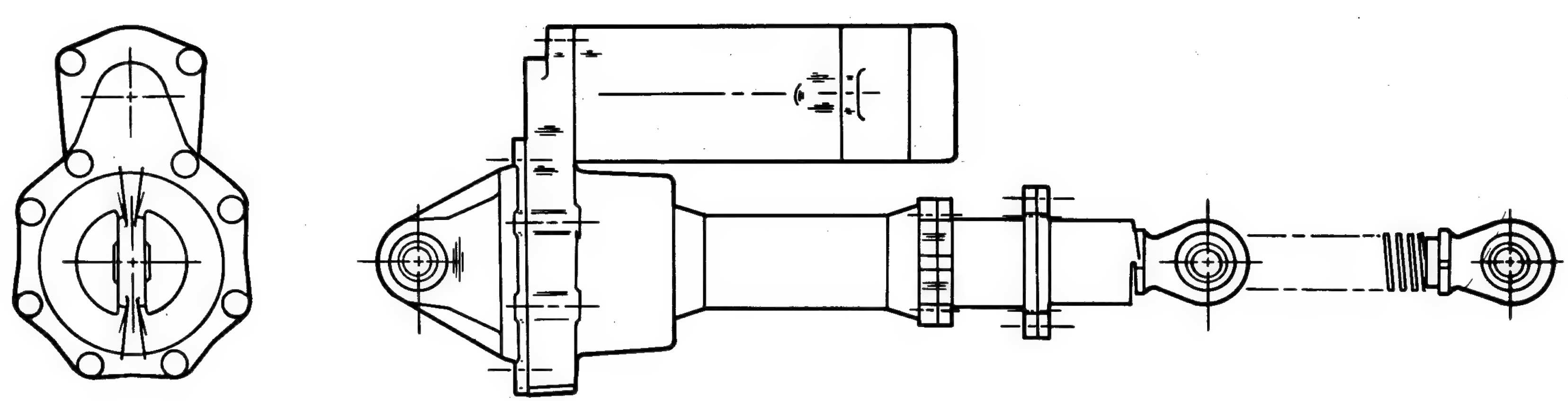


Figure 79. - "Swallowed screw" ball screw design.
(Courtesy Sunstrand Aviation)

6.1.6 Eccentuator. - The eccentuator is a relatively new and interesting approach to the actuation of control surfaces. This actuation method is a development of Vought Aircraft and consists basically of a simple bent beam. The beam has kinematic mechanizations such that the output end (attached to the control surface) moves through an angular deflection in a plane, while the input half is given a compound nutating type of rotation to ensure the planar output motion. Figure 80 is a schematic illustration of the bent beam showing its possible motions with and without nutation. It can be seen in figure 80 that, if a beam with a bend angle of theta is given a rotation of 180 degrees at its input end about its own axis it will generate an output deflection of 2 theta. However, the output end does not move in a plane which would make it impractical for moving a control surface. If a carrier is provided, as shown in the center of figure 81, and the input end of the beam is driven by this carrier, the output end still moves in a circle. However, if the beam is geared to the carrier in such a manner that the beam rotates 180 degrees clockwise as the carrier rotates 180 degrees, counterclockwise, the output motion then becomes planar and the deflection jumps to 4 theta. This is the principle of the eccentuator.

Figure 81 shows the eccentuator as it would be installed to operate a control surface. It shows the control surface's hinge line passing through the spherical bearing at the center of the bent beam. It shows the extremes of control surface deflection and it also shows the beam at mid travel (i.e., neutral for most control surfaces). Figure 82 shows the mechanical advantage characteristics of the device. It shows a mechanical advantage of 9.51 at neutral and infinity at 90 degrees each side of neutral. It should be noted that the plot shown in figure 82 is based on a beam bend angle (theta) of 6 degrees. Studies have shown that the maximum practical bend angle is 7.5 degrees and that 6 degrees is more nearly an optimum. Without going into a detailed analysis, the reason that six degrees is about optimum can be understood intuitively. At neutral (trail), the air loads are essentially zero and the mechanical advantage is 9.51, i.e., higher than it needs to be. At full deflection the air loads are maximum and the mechanical advantage is infinite; also higher than it needs to be. It is apparent that at some point in between the air loads and hinge moments match. For the 6 degree beam bend angle this point is about 60 degrees each side of neutral or at a mechanical advantage of 19.02. As the bend angle is increased the neutral mechanical advantage decreases quite rapidly (a 7.5 beam bend angle drops the mechanical advantage at neutral to 7.6) and all mechanical advantage values each side of neutral decrease proportionately. In the case of a 7.5 degree beam bend angle the matching point would move in to 45 degrees each side of neutral and the mechanical advantage would drop to approximately 10. Thus, a 1.5 degree increase in bend angle above optimum would cut the effective mechanical advantage nearly in half.

An optimally designed eccentuator (i.e., with a 6 degree beam bend angle) has several major advantages over other actuation approaches as follows:

- It is potentially 81 times stiffer than a corresponding hydraulic actuator.

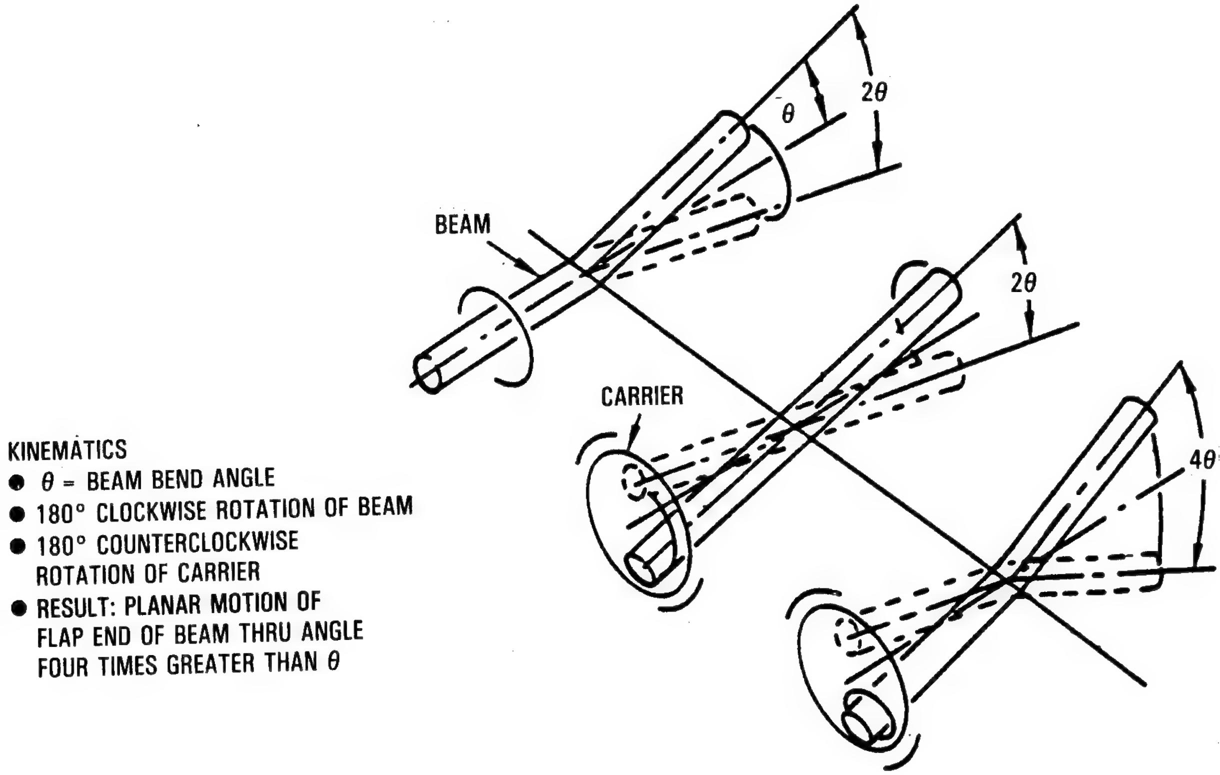


Figure 80. - Operating principle of eccentuator.

- It will adapt to thin wings at least as well as a hingeline actuator and much better than a hydraulic actuator.

- It is potentially the simplest and most reliable actuation approach.

- Its output mechanical advantage characteristics tend to match typical control surface airload curves better than hydraulic actuator kinematics and much better than hingeline actuators.

- It has inherently low inertia and potentially very high frequency response capabilities around neutral.

The eccentuator however has some definite disadvantages relative to other actuation techniques. These are listed as follows:

- Its output deflection, when direct connected to the surface, is low for meeting the deflection requirements of most control surfaces (i.e., 24 degrees for the optimum 6 degree bent beam versus 60 degrees for a typical control surface).

- It will not blowback from the extreme deflected position in the event of a power system failure because of its infinite mechanical advantage at that position; for the same reason, it cannot be back driven by another system.

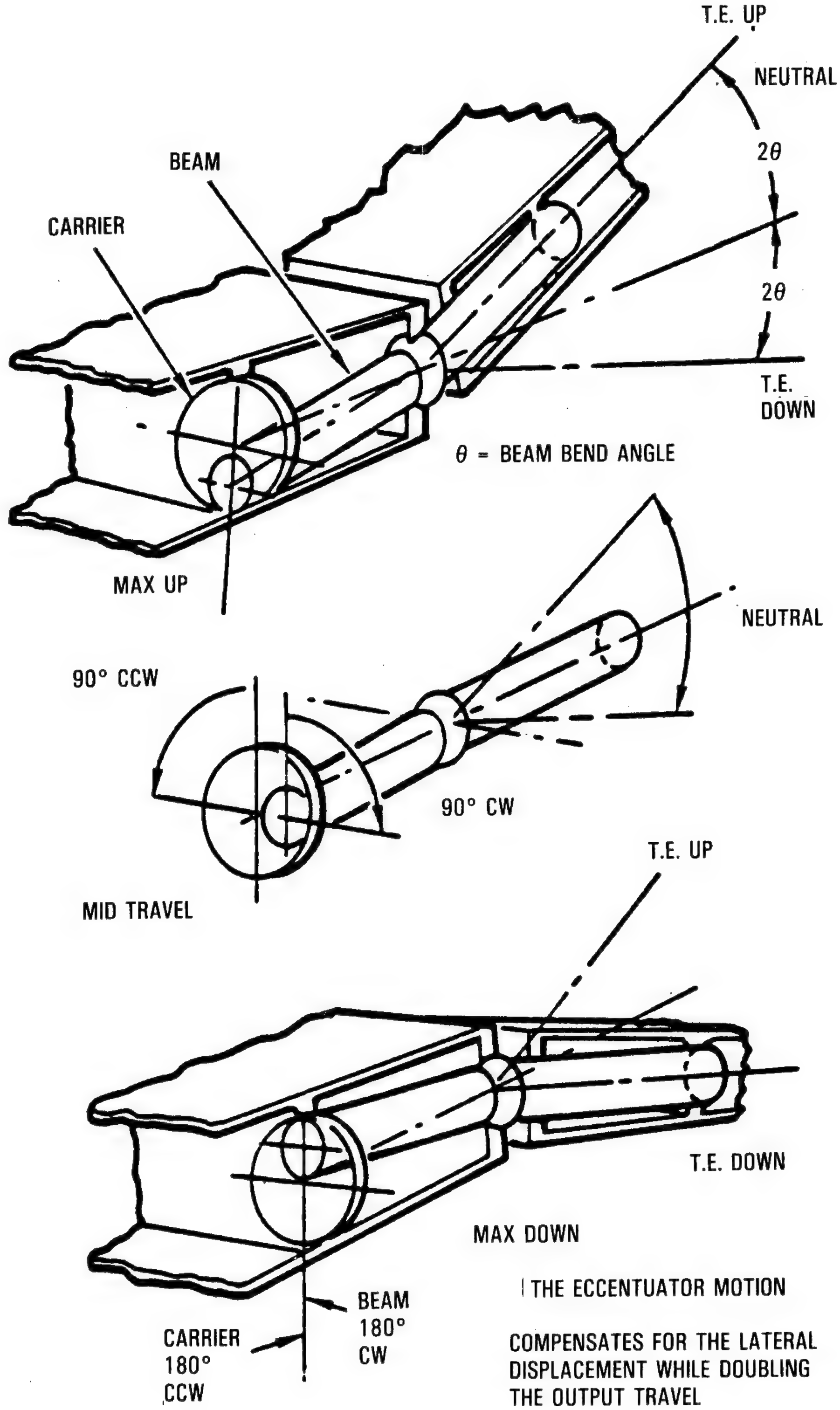


Figure 81. - Wing installation of eccentuator. (Courtesy Vought - LTV).

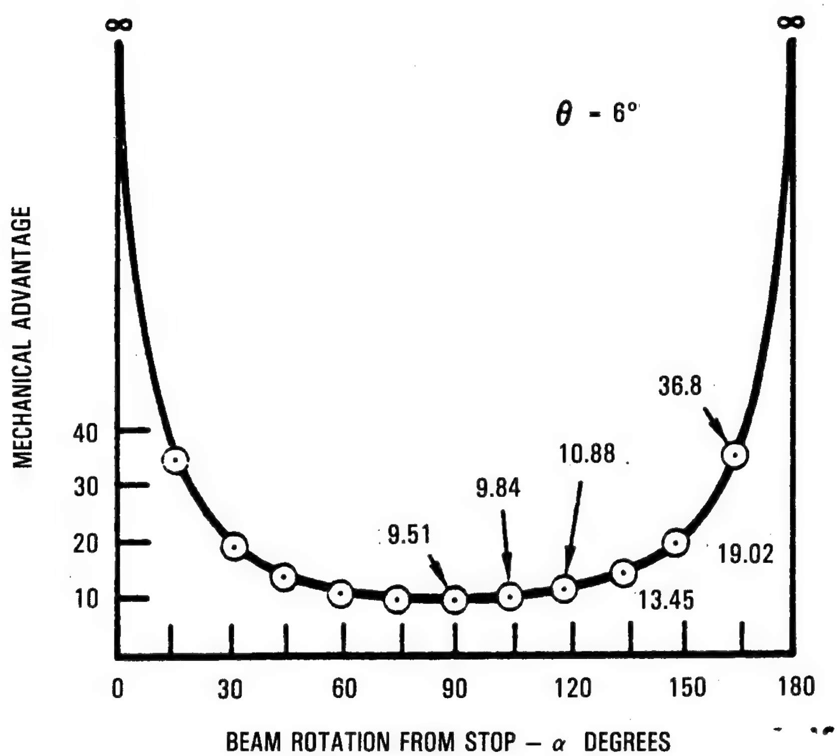


Figure 82. - Inherent mechanical advantage of eccentuator.

- When it operates in the high mechanical advantage area its ability to meet surface rate requirements will be seriously degraded.

- It has little development or operational background behind it.

6.1.7 Infinitely variable traction transmission. - An infinitely variable traction transmission is a mechanical device which transmits power by means of fluid shear in the mechanical interfaces between two highly loaded rolling contact surfaces. Figure 83 shows a simplified schematic of the "differential toroidal" version of this type of transmission. This unit will accept unidirectional input rotation and, using a small steering force on the steerable rollers, it can change the output from approximately half input speed, in one direction, through zero to half input speed, in the other direction. When interposed between between a motor and a hingeline actuator (or a screwjack), the infinitely variable differential toroidal transmission (IVDTT) becomes a mechanical servo. It has seen limited service as a generator constant speed drive and has demonstrated the ability to meet flight control frequency response requirements.

Electrically, the IVDTT has the main advantage that the motor can run at a constant (and very high) speed thereby avoiding the need for the motor rotor-inertia to be constantly accelerated and decelerated from a zero rpm condition. This device also bypasses another nagging problem constantly encountered by EMAS of "holding loads", since when the motor is rotating at essentially a constant speed (when the output is "holding" at zero rpm), it

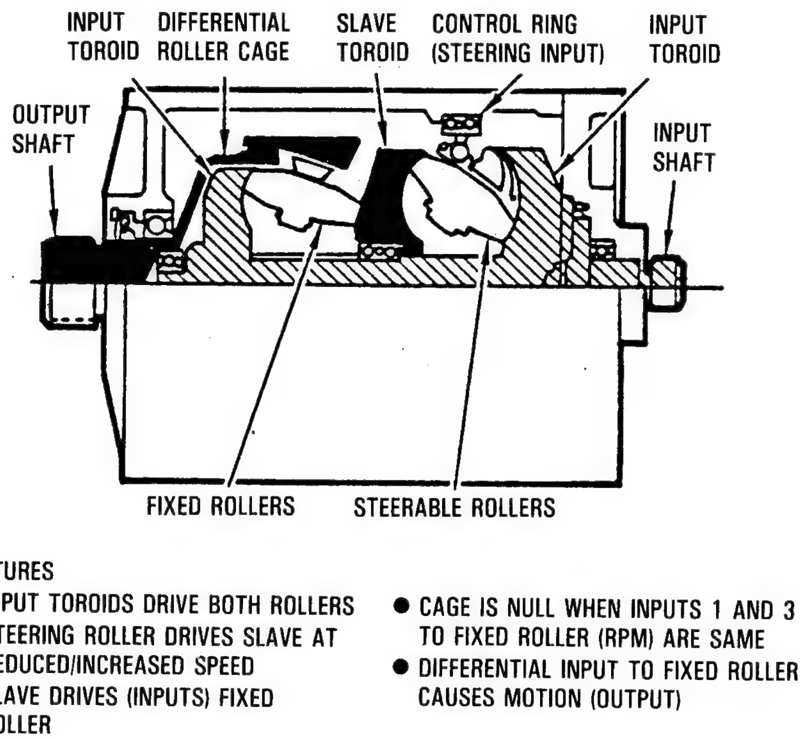


Figure 83. - Traction drive mechanical servo.

sees the holding load as a dynamic load, instead of a stall load. There are not therefore the high current flow and heating problems associated with "stall" operations. The IVDTT is also an inherently high gain stiff transmission. When coupled to the relatively high gear reduction characteristic of hingeline actuators or screwjacks, it essentially becomes independent of output load inertial effects.

In contrast to its many advantages, the IVDTT is a complicated device and it uses numerous high precision parts; more pertinently, it requires extensive development before it can be considered as a viable FCS EMAS.

6.1.8 Dynavector. - The dynavector is in the genealogy of the harmonic drive: to understand the first, one must understand the second. The harmonic drive, shown schematically in figure 84, consists of three primary elements.

- A circular spline (thickwall ring gear)
- A wave generator (an oval rotating element)
- A flexspline (a flexible spline deflected into engagement with the teeth on the circular spline by the wave generator).

The uniqueness of the harmonic drive derives from the fact that the flexspline has one or two less teeth than the ring gear (the circular spline), with which it engages. As a consequence the harmonic drive can effect very large speed reductions in one stage. Thus, with 202 teeth in the circular

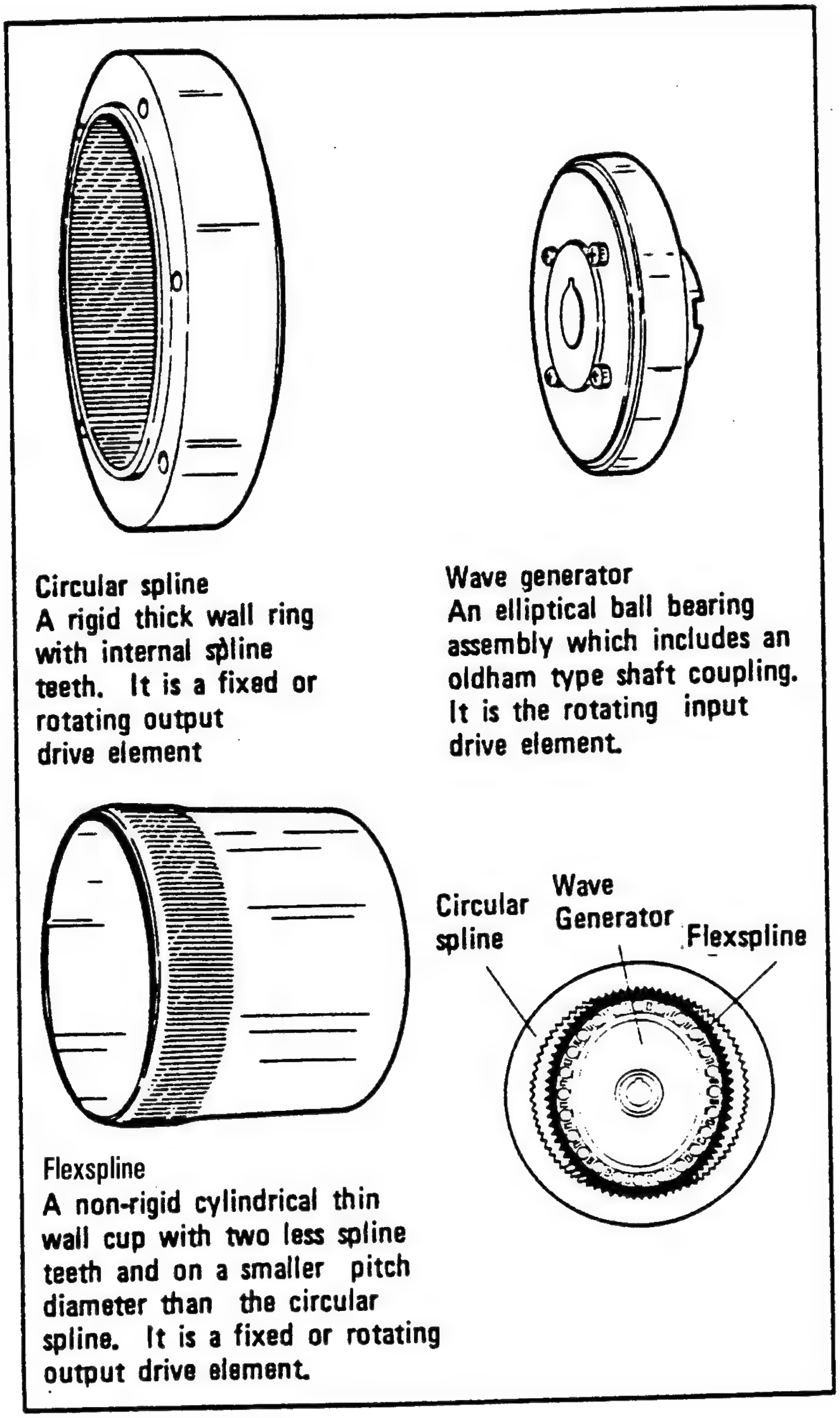


Figure 84. - Harmonic drive.

spline and 200 on the flexspline the ratio is 200/(202-200) = 100:1. In effect, the harmonic drive is a special form of epicyclic gear train in which very large gear reductions are possible in a single stage.

The dynavector is a Bendix development whose design is based on the harmonic drive principle. It uses the epicyclic gear concept to effect direct hingeline-type actuation of a control surface or other loads. Unlike the eccentuator, it is not limited in rotation and it can be adapted to digital (stepper-type) control in that the movement of the output shaft can be effected in discrete increments. The dynavector derives its name from the fact that a diametrical force (a force vector) urges the flexspline into engagement with the circular spline, causing the latter to rotate smoothly, or in small incremental steps, through 360 degrees or more of rotation. The dynavector has been adapted to pneumatic, hydraulic and electrical power sources with a greater portion of the development work being expended on the first two (hydraulic and pneumatic).

Figure 85 is a schematic illustration of the dynavector showing it to be comprised of three basic power transmission elements:

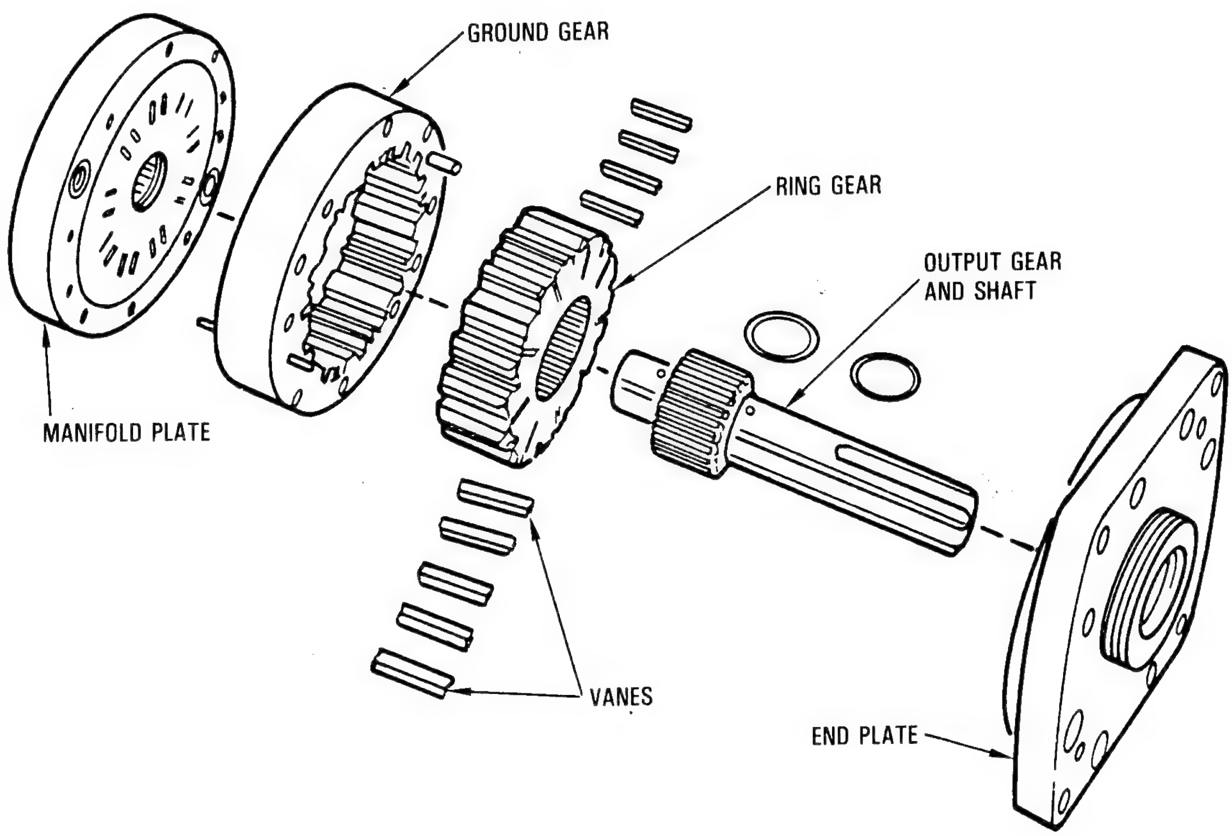


Figure 85. - Dynavector actuator.

- Ring gear
- Ground gear
- Output gear

As an actuation device the dynavector tends to be compact in length, but relatively large in diameter (i.e., pancake design). Figure 86 is a view of the three transmission elements in the assembled state. This particular dynavector is hydraulically operated through the medium of the pressure and exhaust ports located in juxtaposition to "displacement chambers". In operation, a force vector is created by pressurizing the displacement chambers on one side, while venting the displacement chambers on the other side. Rotation is imparted by sequentially pressurizing the vented (exhaust) chamber adjacent to a previously pressurized chamber, while simultaneously venting the diametrically opposite pressurized chamber. This commutating action causes the force vector to rotate through an arc (angular width) equivalent to one displacement chamber. As the force vector rotates, the ring gear rotates at the same speed, driving the output gear and shaft in the opposite direction at a reduced speed and increased mechanical advantage equivalent to the difference in the pitch of the meshing teeth. For a typical 30/32 teeth meshing difference between the ring and output gears, the reduction would be 15:1. At any one time, half of the displacement chambers are pressurized, while the other half are vented. This results in an orbiting-type motion of the ring gear,

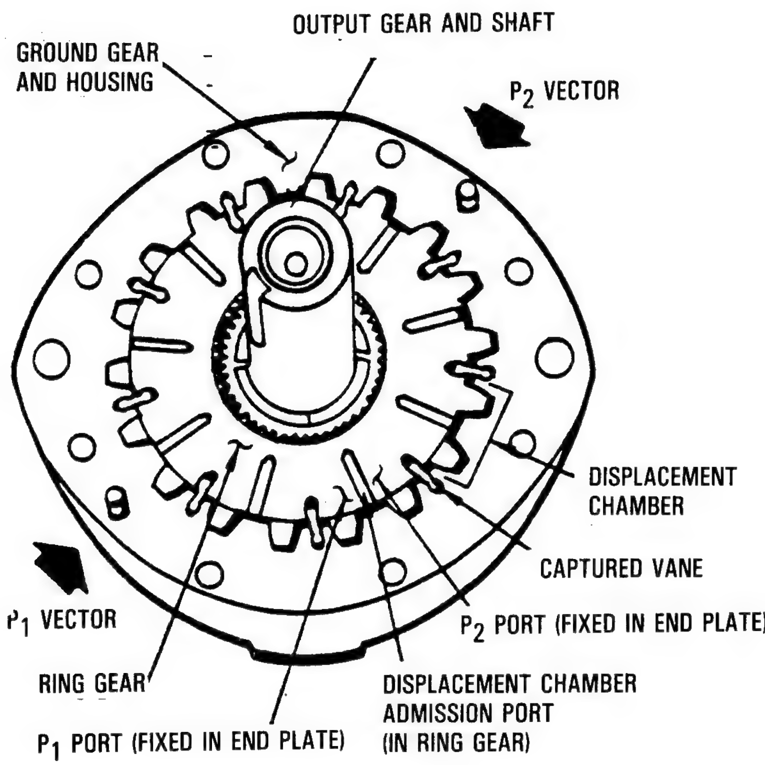


Figure 86. - Dynavector: hydraulically actuated. (Courtesy Bendix)

causing a tooth engagement between the internal teeth on the ring gear with the external teeth on the output gear which is displaced 90 degrees from the force vector.

Although a hydraulic dynavector is not of direct interest to an EMAS based system, an electrical version of a dynavector has also been adapted to produce the rotating force vector. In this case, the force vector is generated by coils, which are disposed circumferentially around the outside of an armature, that is integral with the ring gear. Motion in this type of dynavector is created by commutating the coils in the correct sequence in exactly the same manner as the pressurized/vented chambers were commutated in the hydraulic version. This causes a rotation of the force vector at a rate determined by the commutation rate of the coils. In this electrical version the rotating field can be digitally controlled by sequential energization of the coils, or stepless smooth rotation can be developed by use of a polyphase-winding that produces a sinusoidally changing rotating field. Figure 87 is a schematic of an electrically powered dynavector.

The dynavector is a relatively simple device using, as it does, relatively few parts to get from a raw power input to a controlled power output. It appears to have the following advantages:

- High reliability
- High torque-to-inertia ratio
- High stiffness
- High dynamic performance.

The high reliability derives from the fact that there are no high tooth loadings and the rubbing velocities are approximately one tenth of conventional gears. The high dynamic performance derives from the fact that, unlike an electric motor with high rotor inertia due to high speeds, the dynavector has no mass rotating at high speed.

The disadvantages of the dynavector in its current state of development are that its specific weight (kW/lb) is relatively low and its pancake configuration inhibits its installation in many applications such as thin wings. Also, the power and torque levels that are required by primary flight control surfaces in larger sized transport aircraft are high and somewhat beyond the capability of present dynavector designs. Responding to this, Bendix initiated a new design program incorporating new design approaches to overcome these shortcomings. Unfortunately, as a result of cutbacks this program, as well as all other work on dynavector, was terminated. For these reasons, even though dynavector and similar high torque low speed electrical devices (torquers) may have excellent long term promise, they were not considered viable for the IDEA so they were dropped from further consideration for this study.

6.1.9 <u>Integrated actuator packages</u>. - Integrated actuator packages (IAPs), are self-contained power modules incorporating hydraulic or pneumatic interfaces, but still operating electrically. IAPs can be considered for flight control and surface actuation but they were eliminated from selection for the following reasons:

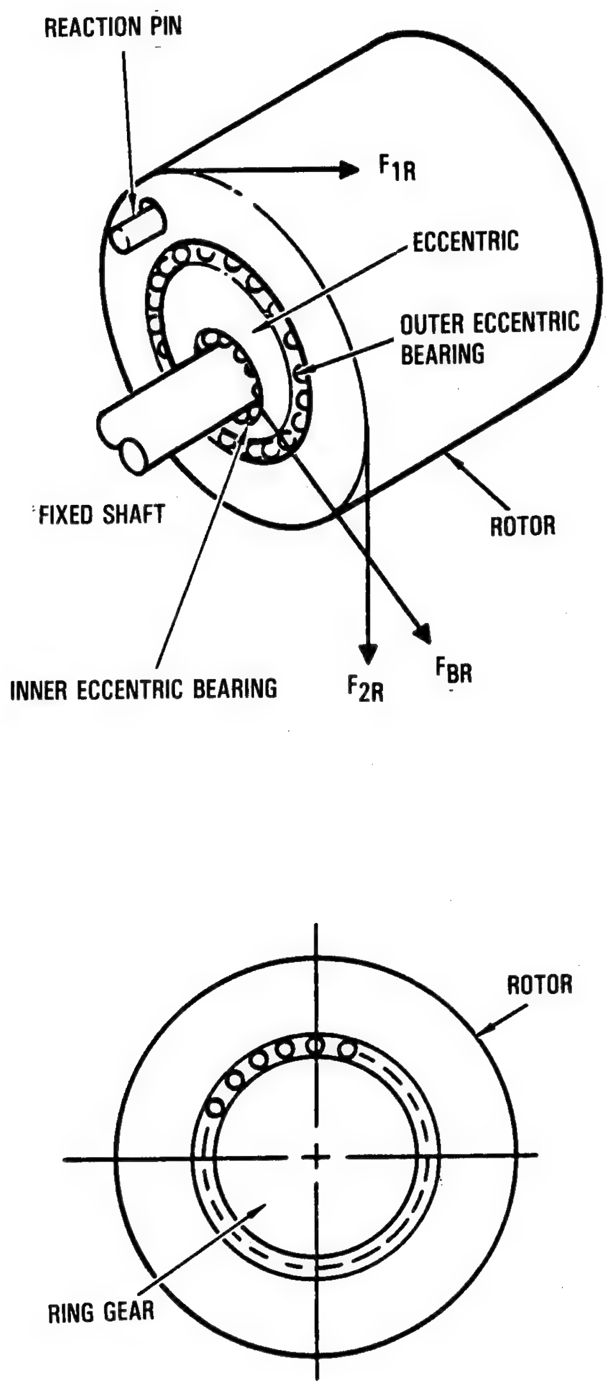


Figure 87. - Dynavector II.

- They were less efficient since an additional different type power interface was always involved.

- They were heavier because of the additional power interface components.

- They had the external appearance of an electromechanical actuator, but internally they employed hydraulics to pneumatics. Thus, in maintenance support hydraulic and/or pneumatic technicians would still be required.

In view of the above, the IAP was not considered as a strong candidate for the FCS EMAS or other applications.

6.1.10 Utility function actuation. - The utility functions considered during the study were as follows:

- Main Landing Gear Retract Actuator
- Main Landing Gear Door Actuator
- Main Landing Gear Door Lock Actuator
- Main Landing Gear Down Lock Actuator
- Main Landing Gear Truck Leveler Actuator
- Main Landing Gear Brakes
- Main Landing Gear Retract Actuator
- Nose Landing Gear Uplock Actuator
- Nose Landing Gear Steering Actuator
- Trailing Edge Flap Actuation
- Leading Edge Flap Actuation
- Thrust Reverser Actuation
- Tail Skid Extension

With the exception of nose wheel steering (which can be a simple open loop servo control), almost all non FCS actuation functions could employ a highly reliable/rugged squirrel cage induction motor as the power source. For landing gear actuation various forms of geared rotary actuators were considered with the rotary actuator powering a trunnion and/or drag-link knuckle. In each instance, however, the geared rotary actuator proved heavier and more demanding of space so the simple expedient use of a screwjack interface appeared more feasible.

The flaps and thrust reverser presented a somewhat different circumstance since the operating mechanism for each of these functions is already powered by a hydraulic or pneumatic motor. Therefore the more simple and direct approach was to use an electric motor to drive a mechanism that was already well proven and developed.

6.2 IDEA: EMAS Applications

6.2.1 Flight control actuation. - Referring to table 11, four categories were eliminated during the preliminary evaluation phase early in the trade study. These were categories 7, 8, 9, and 11.

The following were therefore the viable candidates for the IDEA.

6.2.2 Eccentuator. - While the eccentuator provides many advantages, its main disadvantage was its limited output deflection. To evaluate the deficiency several geometric configurations were considered including one to install two eccentuators in series. This provided additional deflection angles up to plus or minus 48 degrees; also its double hingeline gave an interesting camber to the wing cross section when fully deflected. Driving the second eccentuator and achieving the required nutating action was however a problem that could not be overcome. A second approach was therefore pursued and this appeared very practical. In this case, the surface hingeline was moved aft of the eccentuator's ball pivot as shown in figure 88. This was adapted to the IDEA airplane's outboard aileron chordal cross section at approximately midspan of the aileron where it can be seen that the eccentuator fits nicely within the wing and control surface mold lines. This configuration places the control surface hingeline (aileron pivot point in figure 88) 4 inches aft of eccentuator's ball pivot and give $\pm 20°$ deflection.

Moving the control surface's hingeline aft of the eccentuator's ball pivot does reduce the overall mechanical advantage of the system in a manner somewhat analogous to effect of increasing the eccentuator beam's bend angle, but the end effect is different. Increasing the beam's bend angle to achieve a higher output deflection distorts the mechanical advantage curve so the matching point between load and mechanical advantage moves to a much lower value. However, while moving the hingeline aft results in a higher load on the eccentuator's output, the matching point remains the same in terms of mechanical advantage.

Figure 89 also shows the gearing necessary to achieve the nutating action required to implement the eccentuator principle (see figures 89 and 90). Although not shown in figure 88, the electric motor working through a planetary gear train meshes with the bull gear shown in the figure. This gear performs the function of the carrier shown in figures 89 and 90. As it rotates in one direction it causes the pinion gear, which is constrained by the reaction ring gear, to rotate in the opposite direction. The pinion gear is keyed to the eccentuator beam and thus, the beam is forced to rotate in the counter rotating manner necessary to achieve the double deflection planar motion of the output. The torque at the bull gear input can be very low since there is a gear reduction/mechanical advantage of approximately 200:1.

Figure 89 is a plan view schematic of the outboard aileron installation. It shows the eccentuator bull gear being driven by two independent motor/gearbox units. Each of these motor/gearbox units has its own controller and is powered by a different electrical system. Each motor/gearbox unit is capable of meeting stall load at the design point and at an incipient motor stall speed of 17,000 rpm. During normal operation the motors share load and,

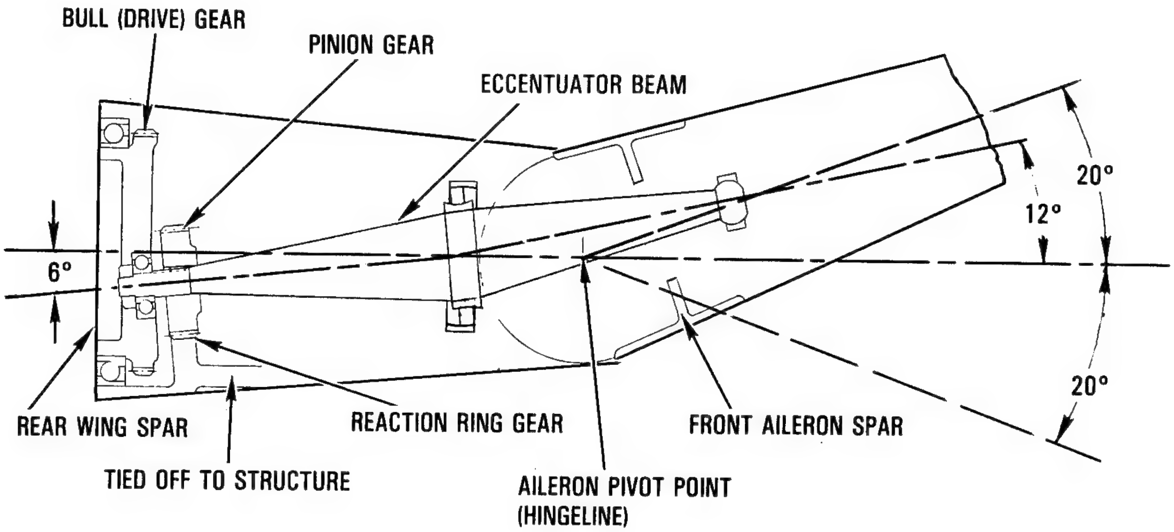


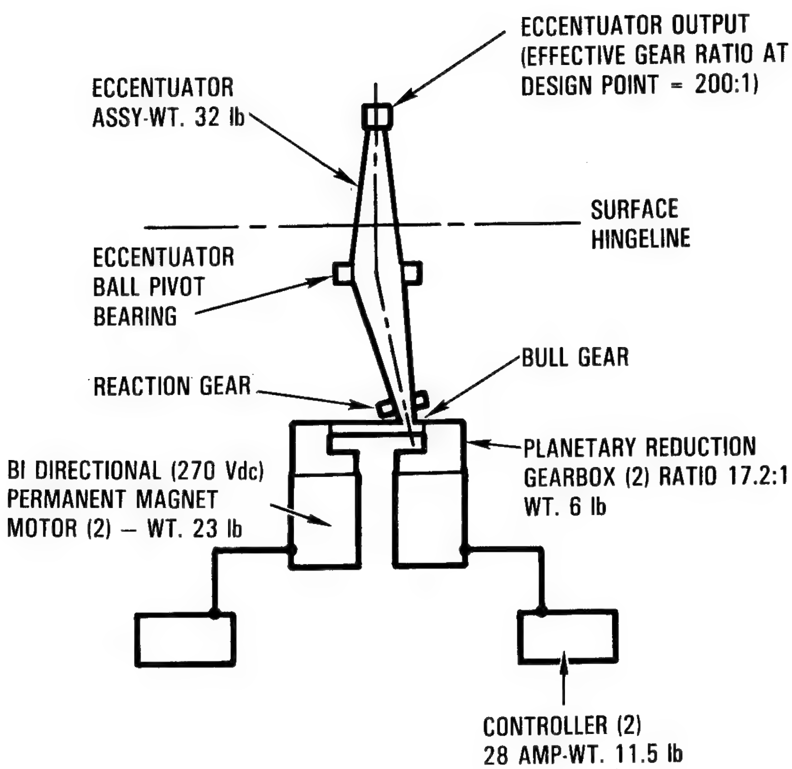
Figure 88. - Eccentuator installation in outboard aileron.

thus are very lightly loaded for most of their operational life. Figure 89 also shows some of the major performance parameters of the eccentuator system and shows a projected weight of the system as 113 lb per surface (226 lb per airplane). This is an excellent system weight, but it is heavier than a hingeline "state of the art" actuation system.

6.2.3 Antijam arrangements. - Originally it was thought jam-tolerant EMAs would prove prohibitive in terms of weight: however, as the study progressed, it appeared this was not the case. Some features of one jam tolerant flight control actuator are shown in table 14.

A rather detailed analysis of the application of jam tolerant actuators to the outboard ailerons was applicable to most other flight control applications. At least three practical approaches emerged. Figure 90 shows the functioning of one of the more favored approaches whose features are listed in table 14. A cam and push rod move the surface in much the same manner as a conventional hydraulic actuator except that the cam can be operated by a central gear train with the outer gear train locked, or by the outer gear train with the central gear train locked. The "primary" motor normally drives the central gear train but should it jam electrical logic would lock the brake on the primary motor and the back-up motor would drive the surface.

In this configuration there are only three points where a jam can occur that will lock the whole actuator: the cam and rod end bearings at each end of the pushrod and the bearings which isolate the cam and gear trains from the fixed housing. The probability of failure in the cam and pushrod end bearings was considered about equal to that of the rod and head end bearings of a conventional hydromechanical actuator. Similarly the probability of a jam in



*TORQUE AT BULL GEAR = 87,600 in-lb/200 RATIO/.85 EFF.
= 515.29 in-lb

*TORQUE AT MOTOR PAD = 515.29 in-lb/17.14 RATIO/.94 EFF.
= 31.98 in-lb

$$\text{MOTOR POWER} = \frac{31.98 \text{ in-lb} \times 17{,}000 \text{ rpm}^{**}}{84483.9} = 6.44 \text{ KW OUTPUT}$$

INPUT CURRENT = 6.44 kW/.85 EFF./.27 = 28.06 amp

TOTAL WEIGHT = 32 + 2(6 + 23 + 11.5) = 113 lb

*INCIPIENT STALL-BEAM ROTATED ±60° FROM NEUTRAL
**INCIPIENT STALL MOTOR SPEED

Figure 89. - Eccentuator schematic and characteristics (outboard aileron).

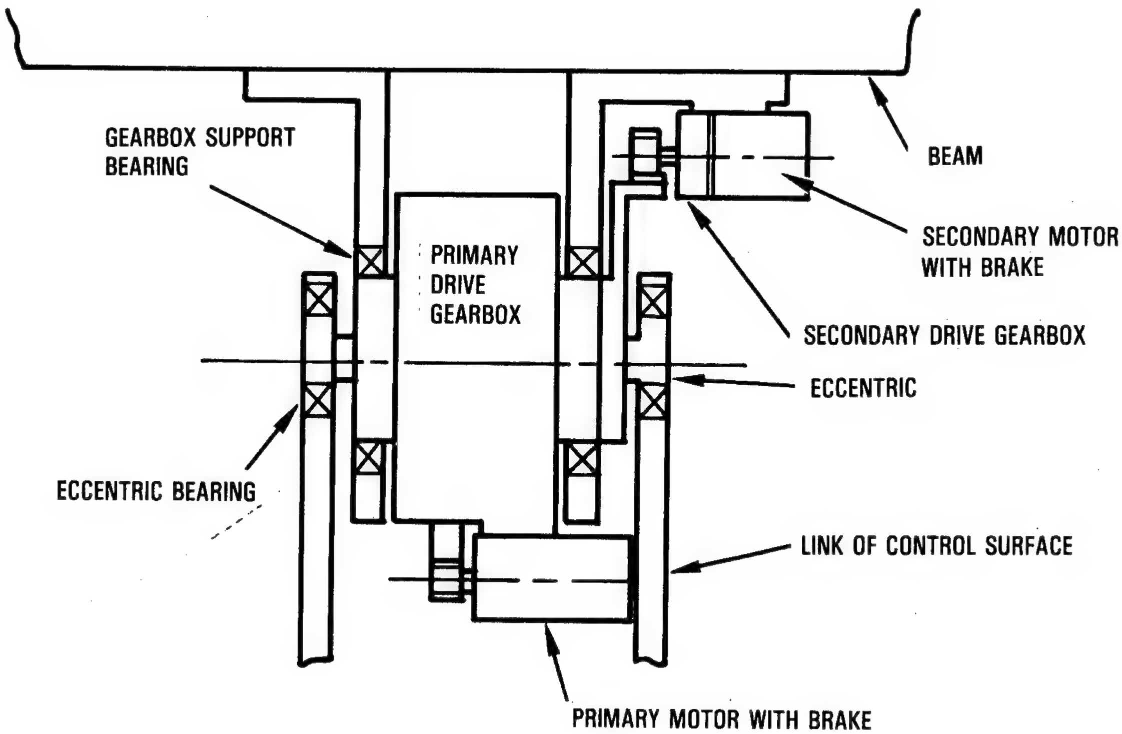


Figure 90. - FCS EMAS candidate with antijam features.

the fixed housing support bearings was considered to approximate the probability of a jam in the piston of rod areas of a conventional hydromechanical actuator.

Figure 91 shows an installation envelope of an antijam arrangement designed specifically for the outboard aileron of the IDEA airplane by MPC Products Corp. This envelope is based on a detailed design and analysis which gave it a high credibility and provided a basis for sizing and evaluating its application to the spoilers, the rudder, and the inboard ailerons as shown in table 15. It can be seen in figure 91 that the unit fits easily in the space available behind the rear spar and that it operates the aileron using the existing surface hingelines and aileron arms. A detailed analysis of the four bar linkage formed by the eccentric, the pushrod, and the aileron lever arm is shown in table 15. It shows the instantaneous torques, lever arms, and link loads at various surface positions and shows that the mechanical advantage of the linkage system matches the aerodynamic load characteristics of the surface.

The antijam mechanization shown in figure 91 has a number of distinct advantages relative to conventional hydromechanical or hingeline actuators. The first is that it uses a relatively large diameter eccentric bearing on

TABLE 14. - FLIGHT CONTROL EMAS CHARACTERISTICS - ANTIJAM

PARAMETERS / ACTUATION FUNCTIONS	FLIGHT CONTROL FUNCTIONS									
	SPOILERS									
	1	2	3	4	5	6	INBOARD AILERON ACT.	OUTBOARD AILERON ACT.	RUDDER ACT.	HORIZ. STAB. ACT.
SURFACE/SYSTEM PERFORMANCE										
SURFACE RATE (DESIGN) (DPS)	60	60	60	60	60	60	35	35	30	7.3
LOAD (DESIGN) (FT-LB)	6500	2500	2200	1420	1500	860	4000	4700	7200	20000
STALL LOAD (FT-LB)	9800	4700	4200	2700	3200	1900	5900	7300	9700	80200
POWER REQUIRED @ SURFACE (KW)	9.23	3.55	3.12	2.02	2.13	1.22	3.31	3.89	5.11	3.46
BANDWIDTH @ +/- 0.1 DEG -3 db (HZ)	3.2	3.2	3.2	3.2	3.2	3.2	3.2	2.2	1.6	3.9
AMPLITUDE (DEG)	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1.5	1	.3
ACTUATOR (OUTPUT DRIVE)	1	1	1	1	1	1	3	2	3	4
NUMBER REQUIRED PER SURFACE	1	1	1	1	1	1	1	1	1	4
NUMBER REQUIRED PER AIRCRAFT	2	2	2	2	2	2	2	2	1	4
TYPE	AJA	AJA	AJA	AJA	AJA	AJA	AJA	AJA	AJA	SJ
EFFICIENCY (%)	55.1	55.1	55.1	55.1	55.1	55.1	54	54	53	82.5
GEAR RATIO (X:1)	1324.68	1324.68	1324.68	1324.68	1324.68	1324.68	2271.83	2271.83	2649.36	9810
WEIGHT (LB) PER SURFACE	56.20	25.00	25.00	15.60	15.60	15.60	56.20	60.30	93.80	303.60
MOTOR										
NUMBER REQUIRED PER SURFACE	2	2	2	2	2	2	3	2	3	4
NUMBER REQUIRED PER AIRCRAFT	4	4	4	4	4	4	6	4	3	4
TYPE	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO
SPEED (RPM) NO LOAD	23000	23000	23000	23000	23000	23000	23000	23000	23000	23000
SPEED (RPM) DESIGN	20000	20000	20000	20000	20000	20000	20000	20000	20000	20000
DESIGN OUTPUT POWER (KW)	16.76	6.44	5.67	3.66	3.87	2.22	6.14	7.21	9.65	4.19
MAXIMUM POWER (KW)	25.26	12.12	10.83	6.96	8.25	4.90	9.05	11.20	13.00	16.80
CURRENT LIMIT PER SURFACE (AMP)	93.57	44.87	40.10	25.78	30.55	18.14	33.53	41.49	48.14	62.22
WEIGHT (LB) PER SURFACE	14.20	14.20	14.20	7.80	7.80	7.80	21.30	14.20	39.60	56.80
CONTROLLER										
TYPE										
CURRENT RATING (AMP)	93.57	44.87	40.10	25.78	30.55	18.14	33.53	41.49	48.14	62.22
NUMBER REQUIRED PER SURFACE	2	2	2	2	2	2	3	2	3	4
NUMBER REQUIRED PER AIRCRAFT	4	4	4	4	4	4	6	4	3	4
VOLUME (CUBIC INCHES										
WEIGHT (LB) PER SURFACE	12.00	12.00	12.00	8.20	8.20	8.20	18.00	12.00	28.50	48.00
TOTAL WEIGHT OF FUNCTION PER A/C	164.80	102.40	102.40	63.20	63.20	63.20	191.00	173.00	161.90	408.40

GRAND TOTAL 1380.65

Acronyms

AJA	Anti-Jam Type Actuator
SJ	Ball Screwjack Actuator
REPM IO	Rare Earth Permanent Magnet Motor - Inside Out Design

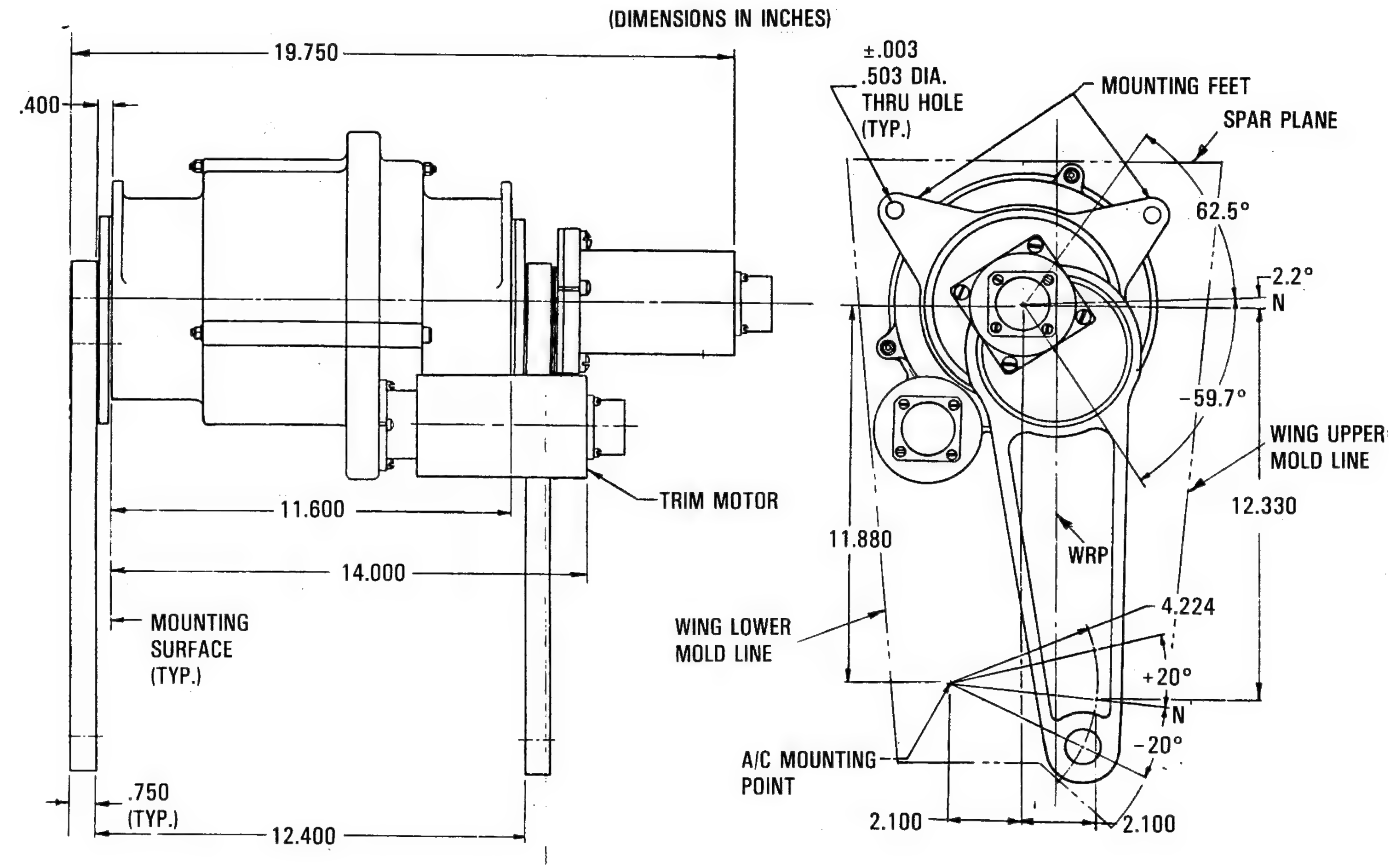
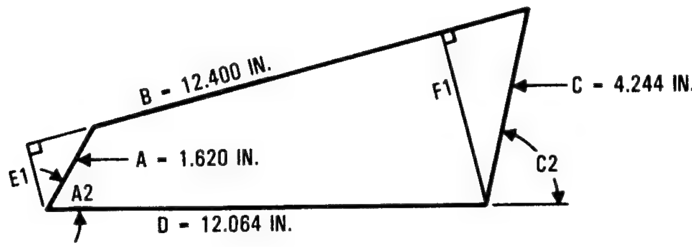


Figure 91. - Outboard aileron antijam actuator envelope. (Courtesy MPC Products)

TABLE 15. FOUR BAR LINKAGE FOR OUTBOARD AILERON ACTUATION.



θ Acft. Surface Angle (deg)	C2 (deg)	A2 (deg)	T2 Torque On Surface (lb-in.)	T1 Torque From Actuator (lb-in.)	Act. Effective Lever Arm E1 (in.)	Surface Effective Lever Arm (in.)	Load On Link B (lb)
-20	73.909	42.513	87,600	18,983	0.778	3.661	24,169
-18	75.	50.887	77,294	20,796	0.989	3.752	20,806
-16	77.	57.973	66,988	20,486	1.148	3.830	17,666
-14	79.	64.351	56,682	18,884	1.273	3.898	14,688
-12	81.	70.285	46,376	16,411	1.372	3.957	11,837
-10	83.	75.922	36,071	13,328	1.452	4.010	9,087
-8	85.	81.356	25,765	9,817	1.514	4.055	6,418
-6	87.	86.652	15,459	6,014	1.561	4.095	3,814
-4	89.	91.861	5,153	2,030	1.594	4.128	1,261
-3	90.	94.446	0	0	1.605	4.143	0
-2	91.	97.024	3,809	3,809	1.614	4.156	926
0	93.	102.18	11,426	4,519	1.620	4.179	2,762
+2	95.	107.35	19,043	7,471	1.614	4.197	4,583
+4	97.	112.59	26,661	10,305	1.595	4.210	6,396
+6	99.	117.93	34,278	12,959	1.563	4.219	8,207
+8	101.	123.40	41,896	15,364	1.518	4.223	10,020
+10	103.	129.08	49,513	17,441	1.458	4.224	11,841
+12	105.	135.02	57,130	19,095	1.383	4.221	13,673
+14	107.	[illegible]41.33	64,748	20,203	1.289	4.214	15,519
+16	109.	148.16	72,365	20,595	1.173	4.206	17,381
+18	111.	155.78	79,983	20,016	1.029	4.195	19,258
+20	113.909	164.72	87,600	18,006	0.843	4.185	21,145

MAX. ACTUATOR TORQUE @ ABOUT θ = -18° = 20,800 lb-in. (APPROX.)

MAX. LINK LOAD @ ABOUT θ = -20° = 24,200 lb (APPROX.)

W_2 = 5.833 rpm (35 deg/sec), NOMINAL NO LOAD SPEED OF ACFT. SURFACE

(F1/E1) AVE = 3.00, AVERAGE RATIO FROM θ = -20° TO +20°

W_1 AVE = (F1/E1) W_2 = 17.5 rpm (105 DEC/SEC), AVERAGE NO LOAD SPEED OF ACTUATOR

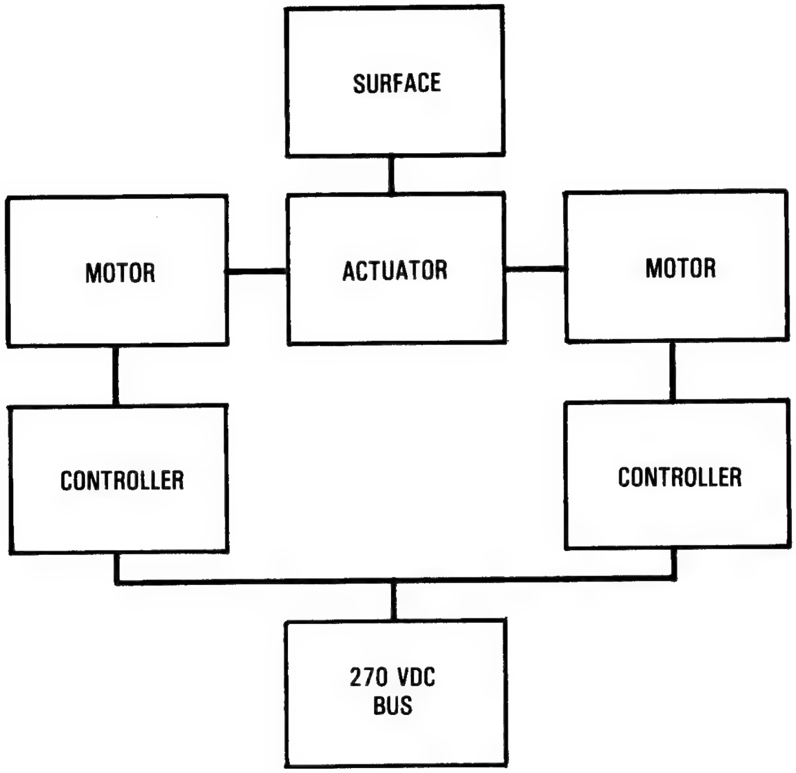


Figure 92. - Schematic arrangement for inboard and outboard ailerons, and rudder actuator systems.

each end of the gearbox to translate output rotation into dual redundant control surface inputs. The electric motor, which mounts to the eccentric, drives the gearbox with an input shaft that is encompassed by the eccentric bearing. This dictates a large sized bearing and the use of such a large bearing enhances gearbox integrity by minimizing bearing stresses and ensuring high stiffness. Second, most all primary control surfaces can be driven by adjusting the eccentric throw and rotation to accommodate the requirements of a particular control surface.

An inherent feature of this configuration is that the over-center toggle action of the EMA is ideally suited to holding continous steady state loads as in the case of the spoilers where there is a "sucking" hinge moment trying to deploy them during flight. Also, it ameliorates a problem endemic to hingeline actuators which react continual type low amplitude/high frequency vibrations. The Electro-Mechanical Power Hinge Actuator (EPHA) avoids this problem by the presence of at least a 3:1 ratio between the eccentric and the surface rotation. This minimizes local fretting wear and backlash. With the exception of the horizontal stabilizer, using linear actuators, all FCS actuators are candidates for anti-jam features of the type described.

In addition to the support provided by MPC Products Corp., Sundstrand Aviation Division of Sundstrand Corp. also supplied assistance. Sundstrand's approach was built around planetary hingeline actuators similar to those which

have been extensively developed by AiResearch. In Sundstrand's approach, however, dual load paths (inverter, cable, motor and input shaft) from a single power source were provided into each end of each hingeline actuation unit (see figure 92). No more than one hingeline power unit was powered by a given power source. Each load path was capable of meeting 100 percent load requirement continuously at somewhat reduced life. Under normal load conditions with less than 50 percent load on each load path, and most generally less than 12 percent, the potential life and reliability of each load path was extremely high. Using the Sundstrand approach as a basis and modifying it through the addition of a package of torque sensors, dog clutches, and brakes, Lockheed succeeded in creating a system which, in most instances equaled or exceeded the reliability (including antijam tolerance) of the antijam systems discussed in the previous paragraphs. This revised system approach was applied to most of the major flight control functions and found to be the most reliable and to have the lightest overall flight control system weight by 131.3 pounds (see tables 14 and 16).

The closest competitor in terms of reliability, weight and cost to the modified Sundstrand load sharing approach, was the antijam approach suggested by MPC. As pointed out in the earlier discussion the antijam approach is not completely jam free, nor disconnect free for that matter, but has about the same jam/disconnect potential as a conventional hydromechanical linear actuator. Since the hingeline actuator load sharing approach has better load path redundancy down to the actuator itself than does the antijam approach and since, in dual and triple redundant applications, the hingeline actuator operates at less than 50 percent duty cycled load compared to the antijam actuator's 100 percent, it was felt that the overall reliability and antijam/disconnect capabilities of the load sharing approach was at least equal to and probably better than the antijam arrangement. For this reason and its weight advantage, the load sharing hingeline approach was selected in the IDEA airplane.

6.3 Identification of Critical Technologies

In common with the concern for the reliability of the electric power distribution systems and the back-up emergency power system for flight critical FCS loads, there must be equal concern for the reliability of the electro-mechanical actuators that will replace the hydromechanical actuators. Evaluation and development of alternative EMAS approaches is recommended to NASA for further technical direction and support.

Specifically, the prospect of actuator jams has been cited as more likely in the electric actuator because of the use of planetary gear reductions and the concern for problems of a broken tooth or a mechanical seizure that could stall a control surface and any actuators connected to it. The present competition for the electric actuator is the hydraulic jack which is a simple power module and one that has intrinsically high reliability. As a consequence, designers have seldom considered the prospects of a hydraulic ram seizure, but there are provisions to bypass the ram in the event of a jammed servo-control valve. One of the few exceptions to this policy might be design approach to the quad-redundant hydraulic jacks on the horizontal stabilizer of Lockheed's L-1011 airplane. Here each of the jacks has a thrust capability to operate the stabilizer, and the stall thrust-capability is such that if one

TABLE 16. - FLIGHT CONTROL EMAS CHARACTERISTICS (HIGH RELIABILITY)

PARAMETERS \ ACTUATION FUNCTIONS	FLIGHT CONTROL FUNCTIONS									
	SPOILERS									
	1	2	3	4	5	6	INBOARD AILERON ACT.	OUTBOARD AILERON ACT.	RUDDER ACT.	HORIZ. STAB. ACT.
SURFACE/SYSTEM PERFORMANCE										
SURFACE RATE (DESIGN) (DPS)	60	60	60	60	60	60	35	35	30	7.3
LOAD (DESIGN) (FT-LB)	6500	2500	2200	1420	1500	860	4000	4700	7200	20000
STALL LOAD (FT-LB)	9800	4700	4200	2700	3200	1900	5900	7300	9700	80200
POWER REQUIRED @ SURFACE (KW)	9.23	3.55	3.12	2.02	2.13	1.22	3.31	3.89	5.11	3.46
ACTUATOR (OUTPUT DRIVE)										
NUMBER REQUIRED PER SURFACE	1	1	1	1	1	1	3	2	3	4
NUMBER REQUIRED PER AIRPLANE	2	2	2	2	2	2	6	4	3	4
TYPE	HLA	HLA	HLA	HLA	HLA	HLA	HLA	HLA	HLA	SJ
EFFICIENCY (%)	77.00	77.00	77.00	77.00	77.00	77.00	77.00	77.00	77.00	89.00
GEAR RATIO (X:1)	2000.00	2000.00	2000.00	2000.00	2000.00	2000.00	3428.57	3428.57	4000.00	16438.36
WEIGHT (LB) PER SURFACE	21.38	10.25	9.16	5.89	6.98	4.15	38.61	31.85	63.48	34.02
MOTOR										
NUMBER REQUIRED PER SURFACE	2	2	2	2	2	2	6	4	6	8
NUMBER REQUIRED PER AIRPLANE	4	4	4	4	4	4	12	8	6	8
TYPE	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO	REPM IO
SPEED (RPM) NO LOAD	23000	23000	23000	23000	23000	23000	23000	23000	23000	21500
SPEED (RPM) DESIGN	20000	20000	20000	20000	20000	20000	20000	20000	20000	20000
SPEED (RPM) CURRENT LIMIT	17126	15675	15562	15592	14688	14392	17253	16949	17751	14742
DESIGN OUTPUT POWER (KW)	11.99	4.61	4.06	2.62	2.77	1.59	4.30	5.06	6.64	3.88
MAXIMUM POWER (KW) (CURRENT LIMIT)	15.48	6.80	6.03	3.88	4.34	2.52	5.48	6.66	7.94	11.48
CURRENT LIMIT PER SURFACE (AMP)	57.33	25.17	22.33	14.38	16.06	9.34	20.28	24.65	29.41	42.51
WEIGHT PER MOTOR (LB)	7.74	3.40	3.01	1.94	2.17	1.26	2.74	3.33	3.97	5.74
WEIGHT PER SURFACE (LB)	15.48	6.80	6.03	3.88	4.34	2.52	16.43	13.31	23.82	45.91
CONTROLLER										
CURRENT RATING (AMP)	57.33	25.17	22.33	14.38	16.06	9.34	20.28	24.65	29.41	42.51
NUMBER REQUIRED PER SURFACE	2	2	2	2	2	2	6	4	6	8
NUMBER REQUIRED PER AIRPLANE	4	4	4	4	4	4	12	8	6	8
WEIGHT PER CONTROLLER (LB)	18.95	10.47	9.60	6.99	7.57	5.12	8.96	10.31	11.71	15.28
WEIGHT PER SURFACE (LB)	37.91	20.93	19.20	13.98	15.13	10.24	53.74	41.24	70.26	122.21
TOTAL WEIGHT OF FUNCTION PER SURF (LB)	74.77	37.98	34.39	23.75	26.45	16.91	108.79	86.41	157.57	202.14
TOTAL WEIGHT OF FUNCTION PER A/C (LB)	149.54	75.96	68.78	47.50	52.90	33.81	217.58	172.82	157.57	202.14

TOTAL FLIGHT CONTROL EMAS WEIGHT 1249.32

Acronyms

HLA Hingeline Actuator
SJ Ball Screwjack Actuator
REPM IO Rare Earth Permanent Magnet Motor - Inside Out Design

jack does jam, the other three can physically break that jack away from its attachment point to structure.

From the above it is clear that mechanical jams in electric actuators can take two forms: address the case where multiple actuators are mechanically-summed onto a control surface, and where a single actuator powers the control surface. In the latter configuration a separate "back-up" motor can be used to control the surface and some novel method must be incorporated to bypass the jam. Such an approach was designed by MPC in the course of this study, and it was described in the EMAS section of this report. The multiple actuator configuration dictates a different and more critical solution, since in this case, one actuator failure can abrogate the concept of redundant actuation of the single control surface. The solution is further complicated by the fact that once an actuator jams, it effectively stalls the other actuators connected to the surface: there must therefore be a nonambiguous method of discriminating, in real time, which actuator is jammed and how to bypass it. One favorable aspect of the multiple actuator approach, however, is that one or more of the other actuators can continue to operate the control surface once the faulty-actuator has been bypassed. This at least avoids the need for a separate backup motor or actuator. Mechanical-summing is preferable, in many cases, to redundant control surfaces. For example, if a split rudder is used on the premise that one section can be stalled in a deployed position and the other section can offset its yaw-moment, then the surface area of the rudder must be increased to provide the same degree of yaw control. On the other hand, the redundant actuator concept does not have this disadvantage.

To summarize, NASA's further support of FCS-EMAS development is recommended and innovative approaches should be encouraged that would permit single and multiple actuators operating critical flight control surfaces to by bypassed in the event of mechanical jams. A corollary of this is that some unambiguous and reliable method should be developed to discern and bypass the faulty actuator where two or more are on a control surface.

7. ICING PROTECTION SYSTEMS

Ice protection in the IDEA involves the engine and wing ice protection systems. Where hot pressurized air was used in the baseline airplane, electric ice protection systems are used in the IDEA. Typically there are two candidate forms of ice protection:

- Electrothermal
- Electroimpulse

The former is the more conventional type of ice protection in which electric heat is applied to the lips of engine cowls and to the leading edge of the wings and/or the empennage. In some large aircraft there is a higher degree of tolerance to ice accretion on the wings and so these aircraft are not so critical of ice protection as are the smaller aircraft. In the L-1011 and the Baseline hot engine bleed air is ducted into the wings and applied through an extensible/retractable tube to the wing leading edge slots. Hot

engine air is also led forward to the engine cowl lips, and a distributor ring around the forward periphery of the engine cowl is used to inject the hot air onto the forward leading edge of the cowl. Figure 93a and figure 93b are typical of this type of ice protection.

For the IDEA an electroimpulse deicing system was selected both for the leading edge slats and the engine cowl lips. This electroimpulse deicing system (EIDS) has its origin in Soviet patents, and some seven western countries, including the U.S., have investigated licensing agreements. The Russian EIDS has been tested in an Ilyushin 18 turboprop commercial airplane and in the Ilyushin 76 turbofan cargo transport. The Lockheed-California Company solicited specific technical data from the USSR on the EIDS in 1972 to permit a more in-depth evaluation of the technology but it was not available, although the Soviet engineers did express a willingness to install EI equipment in any structural test specimen that might be supplied by Lockheed. Contacts were also made with Lucas Aerospace Company in the UK who had worked more closely with the Russian engineers, since this company was seriously considering licensing prospects in that same near time frame. Based on the data available on the Ilyushin aircraft experience and supplementary information from Lucas Aerospace, the Lockheed-California Company initiated its own in-house evaluation program. Several U.S. patents have been issued or are now pending, including some that address the application of the EIDS to composite type structures. Since eddy current conduction is not possible with composite wings, 'driver' (conductive plate) can be attached to the undersurface of the leading edge of the skin to permit eddy current induction. The alternative is a patented system wherein two electromagnetic coils are mounted in close juxtaposition, with one coil attached to the inside of the nonconductive skin and the other to the air structure. These two coils are wound in phase opposition so when subjected to the high discharge currents from the capacitors, the antipolar magnetic fields create a sharp repulsive force between the coils. Some minimal testing was accomplished at Lockheed on this modified EIDS and the results were encouraging. Using a ballistic pendulum as a force monitor the efficacy of the dual-coil approach was verified and it was found that the electromechanical forces produced were superior to the eddy current type EIDS. This superiority resides largely in the efficiency differences of the two systems: the dual-coil method is electromagnetically superior to the basic EIDS approach since the latter relies on the magnetic fields that are set up by the eddy currents. Since this is a scondary induction effect that depends upon the specific resistivity of the metal, the dual coils make a more direct application of the electromagnetic force phenomenon. Tests with the ballistic pendulum shows a 150 percent or more improvement in the dual-coil approach over the basic system. A further advantage is that the total system weight is less, because of the higher efficiency, when the novel approach is compared with an EIDS, using a driver plate.

In a further effort to explore more effective implementations of the EIDS technology, Lockheed pursued the concept of putting torsion into the wing leading edge since this seemed to have the potential for increasing the shear stresses in adhesive bond between the ice and the skin. Again, this was found to be beneficial in the limited testing accomplished.

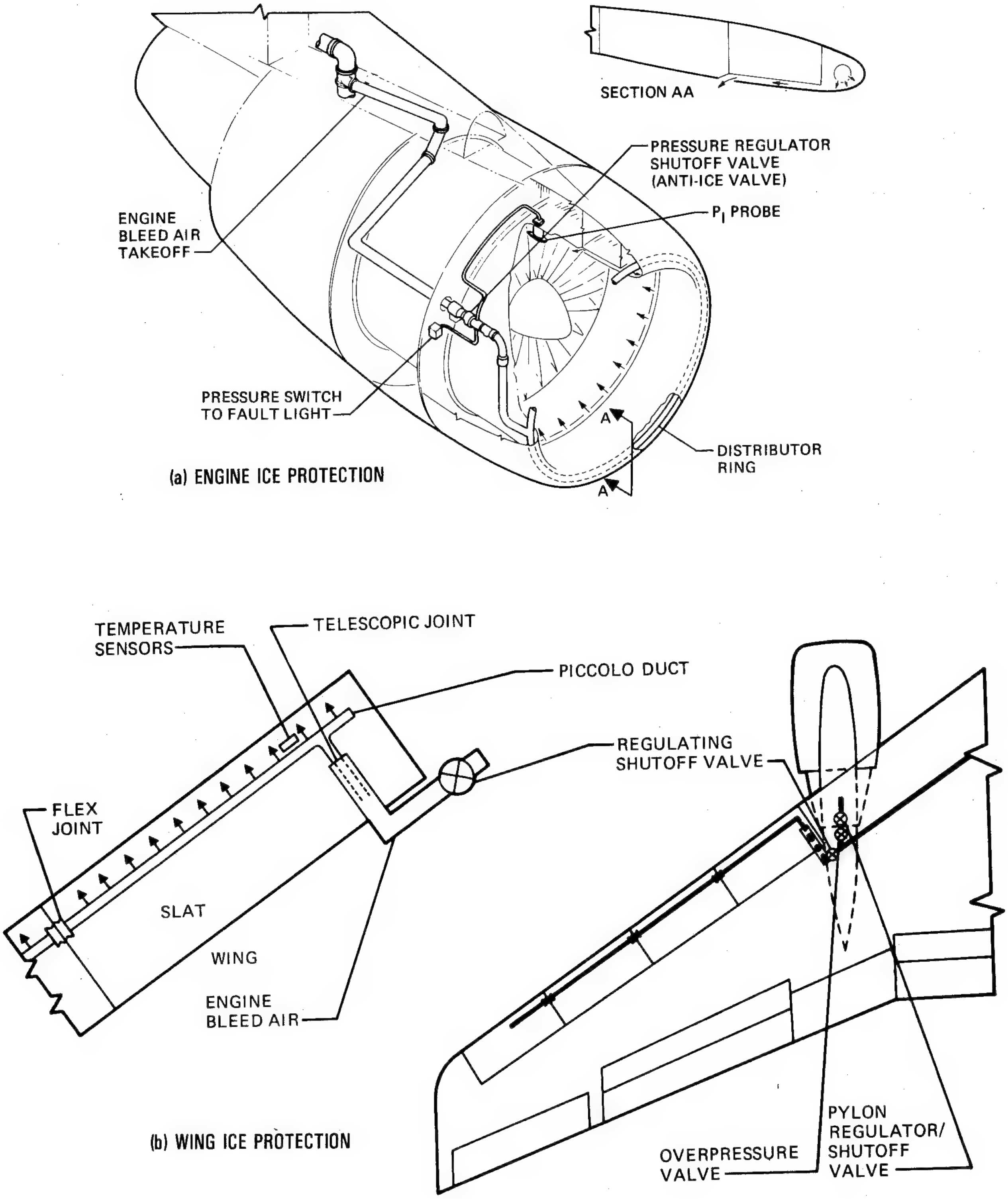


Figure 93. - Hot air wing/engine ice protection.

The principle of the EIDS is based upon the installation of multiple electromagnetic coils located spanwise along the wing leading edge (see figure 94). These coils are rigidly located in proximity to the skin and usually positioned aft of the leading edge on the upper and lower surfaces of the wing leading edge.

In operation, a transient pulse of current is applied to the coils, sequentially under the control of a cyclic timer. During each cycle all of the spanwise coils in each wing are energized, individually or in pairs. The electromechanics of the deicing principle depend upon an elastic deformation principle depend upon an elastic deformation being created in the skin in proximity to the particular coils being energized. More specifically, the very high current pulsed into each coil, sets up strong electromagnetic fields that intersect the skin and induce eddy currents in the skin as a function of the rate of change of flux. The magnetic fields created by these eddy currents then react strongly with the magnetic field of the coils, with the result that a very sharp repulsive force is developed between the coil and the skin. The electromagnetic force has the effect of imparting a hammerless-type blow to the aluminum skin, in the localized area of the coils. As a consequence, the ice appears to "explode" off the surface rather than being shed into the free-stream airflow.

The advantages of the EIDS are many, but the main one is that is an exremely economic way of deicing in terms of the electric power required for a given protected area. Typically this has been stated as about one hundredth of a conventional electrothermal deicing system. More spcifically the Russians' estimate a power of approximately 26 W/meter^2 and this has to some extent been confirmed by Lockheed.

Another attractive aspect of the EIDS is that it is independent of outside air temperature and so the amount of electrical energy required does not have to be increased with very low outside air temperatures. With the electrothermal deicing system the energy imparted to the protected area must be increased as a function of the outside air temperature. This requires 75 to 100 kW for protecting the empennage of a fairly large aircraft, compared to about 1 kW for the EIDS.

7.1 Theoretical Analysis

In the Lockheed EIDS exploratory IRAD programs theoretical analyses were made of the deicing principle since it is not fully understood. The elastic deformation of the skin at the outset induces bending stresses in the ice and causes it to break up into small segments that are precipitated from the skin in a "shower type" pattern. In the process, skin shear stresses are created and a cohesive failure develops in the ice close to the skin or an adhesive failure occurs at the ice/skin interface. In the course of analyzing the process, however, it is clear that the conditions are highly dynamic and complex inasmuch as "ringing" condition is created, resulting in a transverse wave propagation. This is shown simply in figure 95. Evidently these characteristics match a particular set of conditions, such as the amount of energy

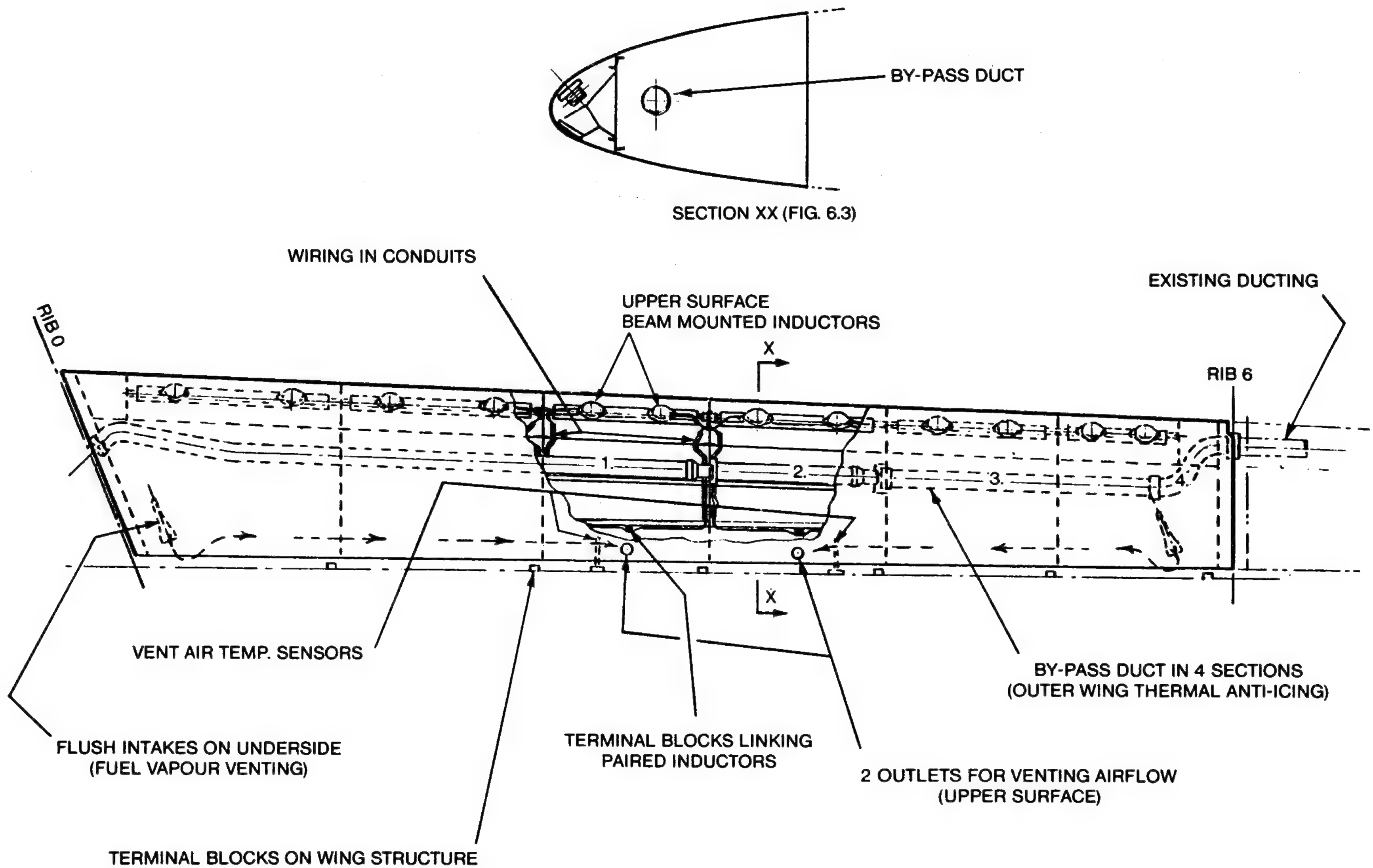


Figure 94. - EIDS: demonstrator aircrafts. (Courtesy British Aerospace)

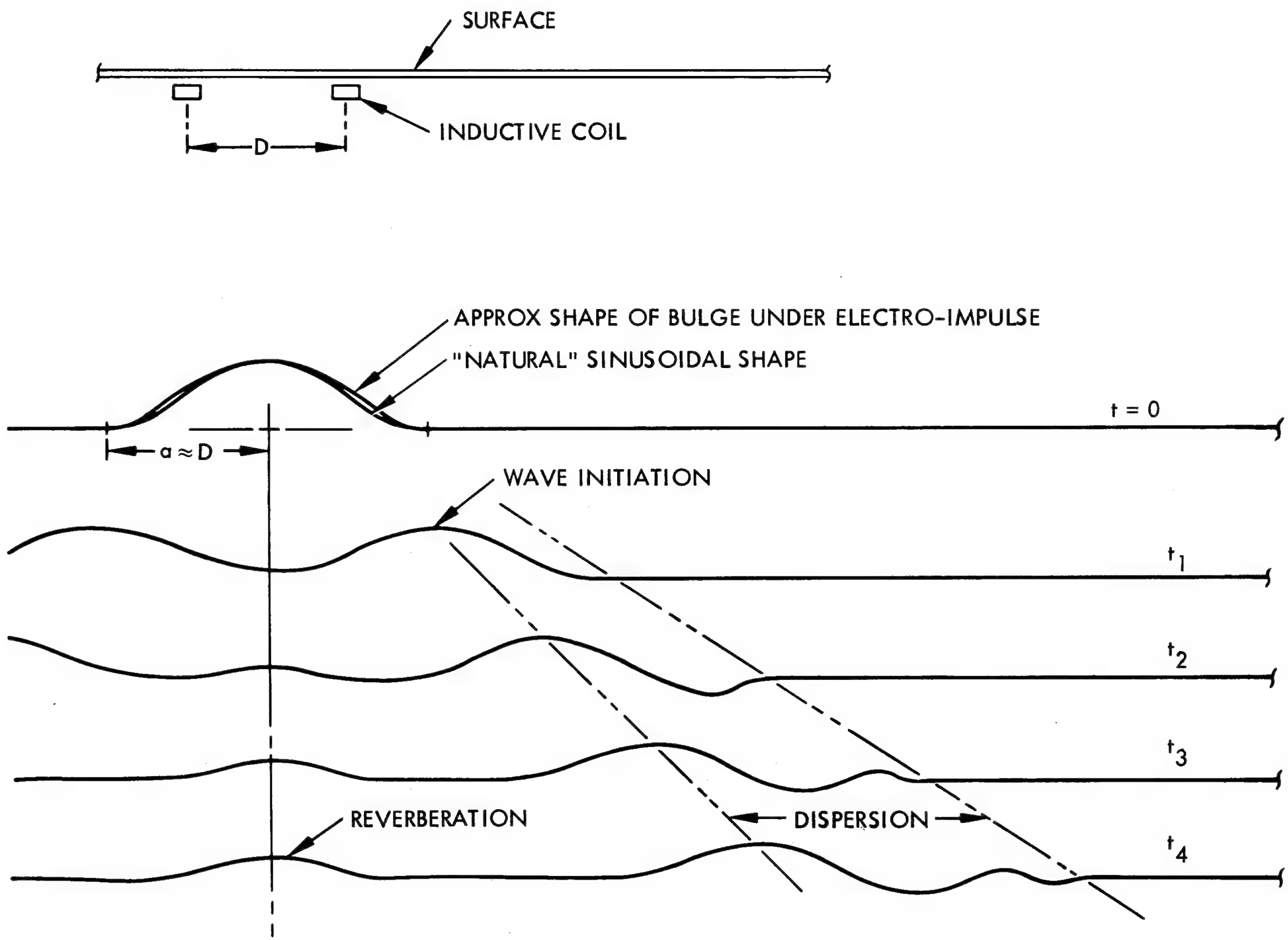


Figure 95. - EIDS: transverse wave propagation.

dumped into the coil, the $d\phi/dt$ of the magnetic flux, the physical characteristics of the skin (electrical conductivity, thickness, rib-spacing, coil spacing) and the amplitude deflection/acceleration rate of the skin (due to electromagnetic forces). Figure 96 shows the theoretical sinusoidal bending mode.

7.2 EIDS Testing

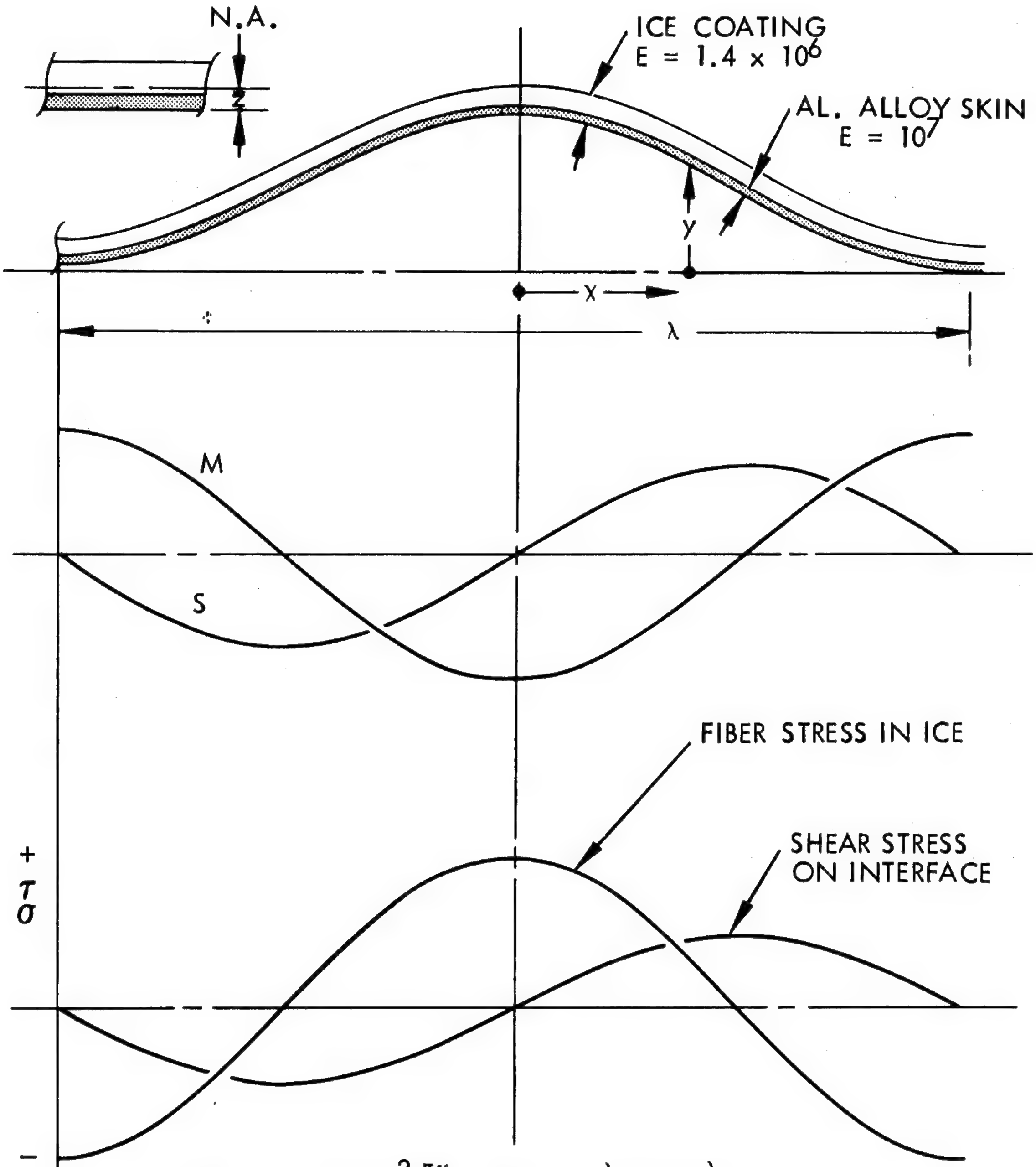
The type of ice formed can also affect the effectiveness of the EIDS, since the ice accretion varies with the outside air temperatures. At temperatures just below freezing the ice accretion can be dense and strong as manifest by the lack of entrained air in the freezing process. In contrast, at low temperatures, because of entrained air, rime ice is deposited which tends to be irregular and this is not as strong nor dense as clear ice. There are, therefore, many interdependent and independent parameters affecting an in depth understanding of the EIDS process so, pursuant to this, tests were conducted on flat panels as well as simulated leading edge sectors. The purpose of some of these tests was to determine whether pure axial strain was effective as deformation/bending wherein shear stresses are developed in the ice/skin interface. The former was found to be ineffective as the axial strain had no observable benefit in terms of reducing the adhesive bond between the ice and skin.

From the electrical standpoint tests were conducted using a variable voltage power supply that was able to vary the applied voltage to the coils up to a 4000 V level. Similarly, the condensers that were charged up by the power supply aried in capacity from 180 μF up 600 μF and more. In each test, the product of the voltage and the capacitor value affected the energy level in accord with the formula ($E=\frac{1}{2}CV^2$). The coils evaluated were typically flat spirally-wound coils (see figure 97) which possessed resistance and inductance. Being air coils, there were no magnetic saturation effects, but the pulse rise-time and the transient ringing frequency were related to the R, L and C values. In the force evaluation tests made on flat panels, and the simulated leading edge sample, the test conditions were varied as follows:

3000 V	50 μF	225 joules
2090 V	180 μF	400 joules
2020 V	391 μF	800 joules
830 V	595 μF	205 joules

Figure 98 is a simple schematic of the laboratory set up for the force measurement tests. Pulse currents into the coil were controlled by a silicon controlled rectifier and the load cell deflections were monitored by the oscilloscope.

Based on the foregoing and the tests conducted on the 30 inch leading-edge specimen (figure 99) in the Lockheed icing tunnel, it was possible to perform a preliminary type design of an EIDS for the IDEA. The leading edge slat configuration and twenty-five coils (approximately 3 inches in diameter) were assumed to be located in each wing along with a centrally-located power supply



$$y = y_{max} \, \text{COS} \, \frac{2\pi x}{\lambda} \qquad \text{for} \quad -\frac{\lambda}{2} < x < \frac{\lambda}{2}$$

$$M = -EI \, \frac{dy^2}{dx^2} = -EI \left(\frac{2\pi}{\lambda}\right)^2 y_{max} \, \text{COS} \, \frac{2\pi x}{\lambda}$$

$$S = \frac{dM}{dx} = EI \left(\frac{2\pi}{\lambda}\right)^3 y_{max} \, \text{SIN} \, \frac{2\pi x}{\lambda}$$

Figure 96 - EIDS: sinusoidal bending relations.

Figure 97. - EIDS: coils and coil suppport structure of leading edge test specimen. (Lockheed).

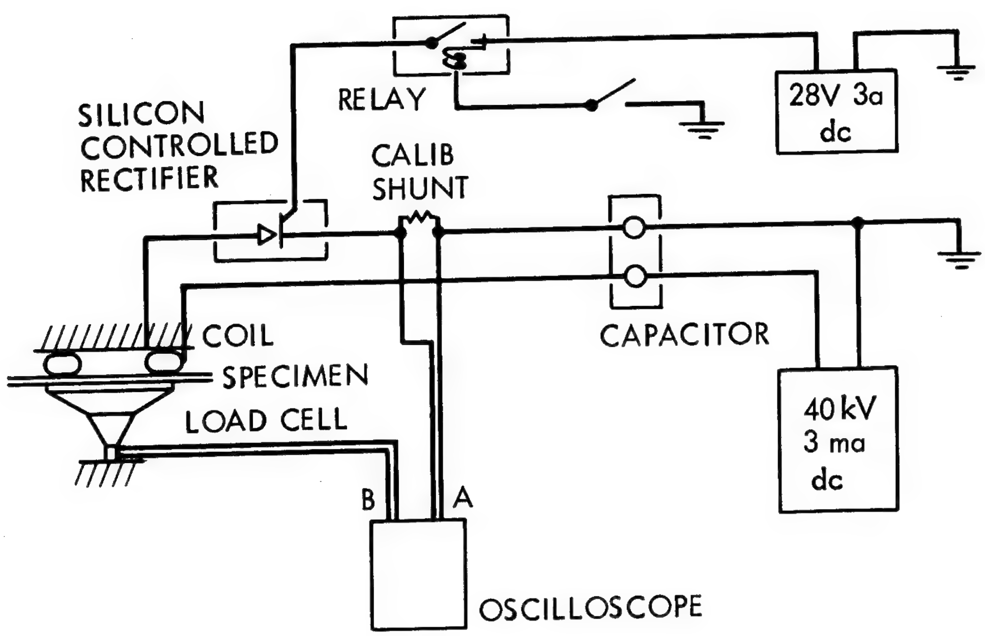
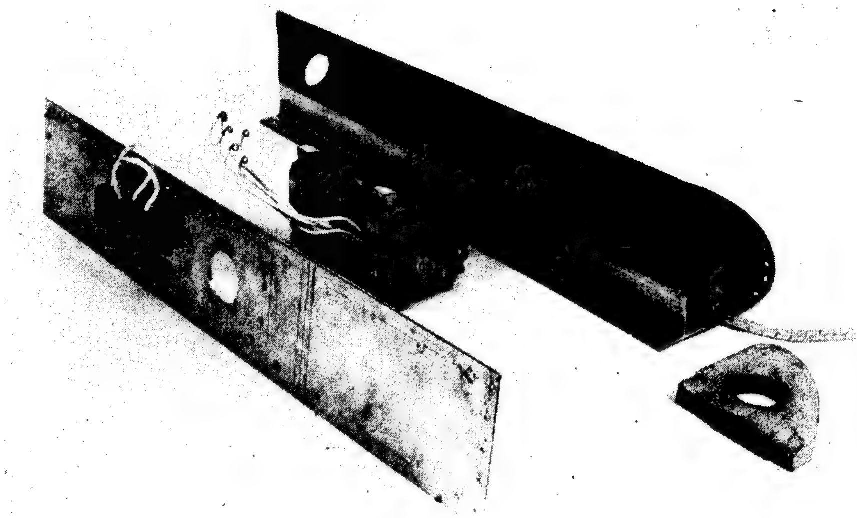


Figure 98. - EIDS: schematic of force measuring tests (Lockheed).



129 450R

Figure 99. - EIDS: leading edge test specimen (Lockheed).

source. Electrolytic capacitors to provide a 800 V 625 μF energy source were also located in each wing for an approximate weight of 4.5 pounds/wing. Other components of the system consisted of the controller and miscellaneous wiring. The estimated weight for the system was about 85 pounds/wing, or a total of 170 pounds. This weight is assessed as being conservative, since it was based primarily on the type of components used in the in-house test program. The average input to each coil was assessed at about 28 watts and the total power per side for the EIDS was estimated at 0.5 kW.

The conclusions to be drawn from the Soviet type EIDS are that the concept is well founded and it has been validated by the installations in the Ilyushin aircraft as well as by flight tests conducted by British Aerospace in a BAC1-11 in simulated icing conditions. In this airplane six coil pairs were installed on the upper and lower surface (24 total) on a 6-foot leading edge test section (see figure 94) on the inner portion of the right wing. Lucas Aerospace furnished a set of USSR supplied equipment to BAe for these tests and a photo-reproduction of the coils and thyristor is shown in figure 100. An inflight photo of the EIDS performance simulated icing conditions in the BAC1-11 is shown in figure 101. Rolls-Royce have also conducted some detailed investigations into the application of the EIDs to fan cowls.

7.3 Plans

The foregoing experience, coupled with the analytical and icing tunnel tests at the Lockheed Burbank facility, validate many of the Russian claims for the EIDS and so the system warrants further NASA support. There are many aspects to the performance of the system and it is necessary to plan for a

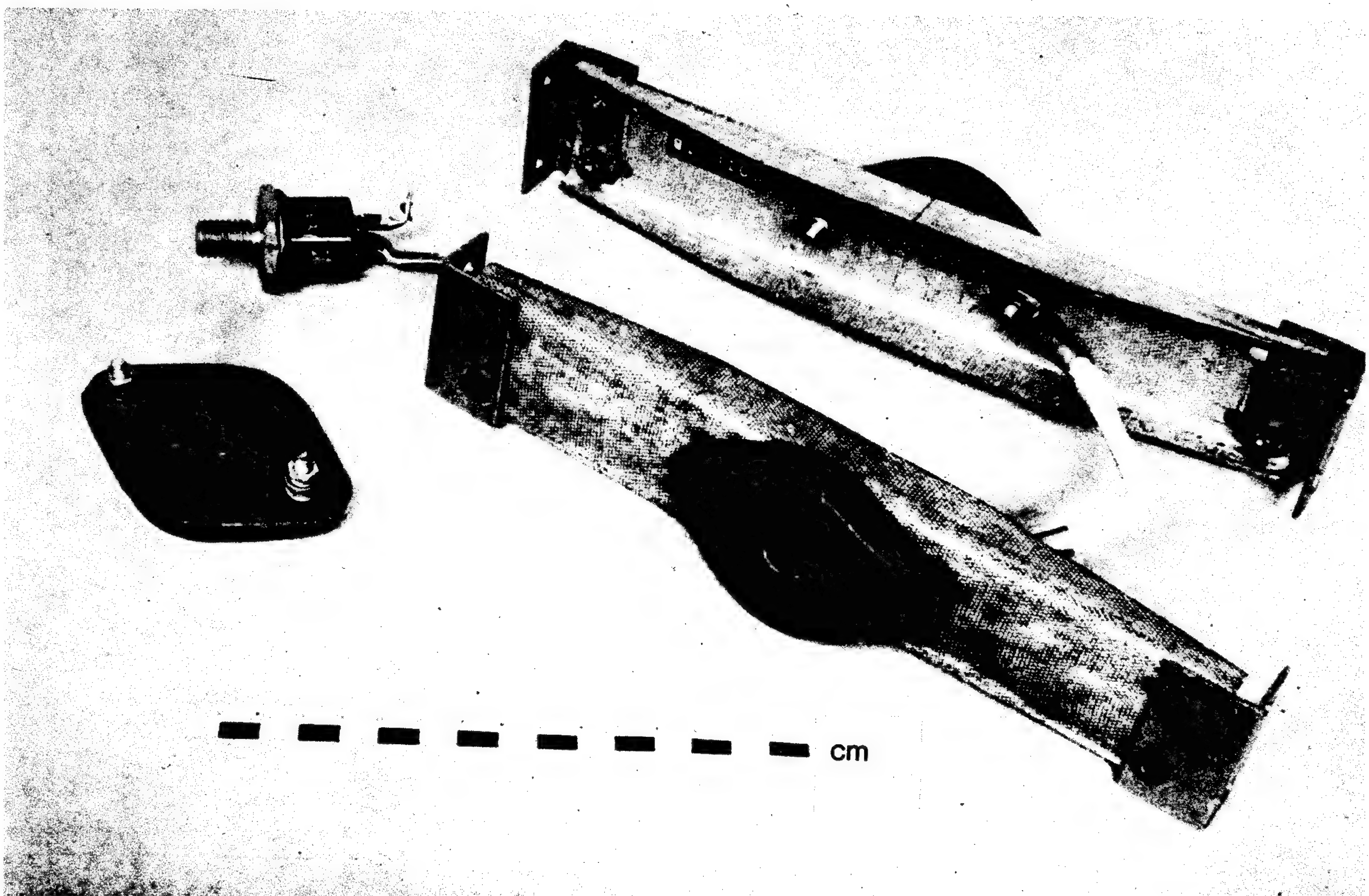


Figure 100. - EIDS: USSR produced coil and thyristor. (Courtesy British Aerospace/Lucas Aerospace)

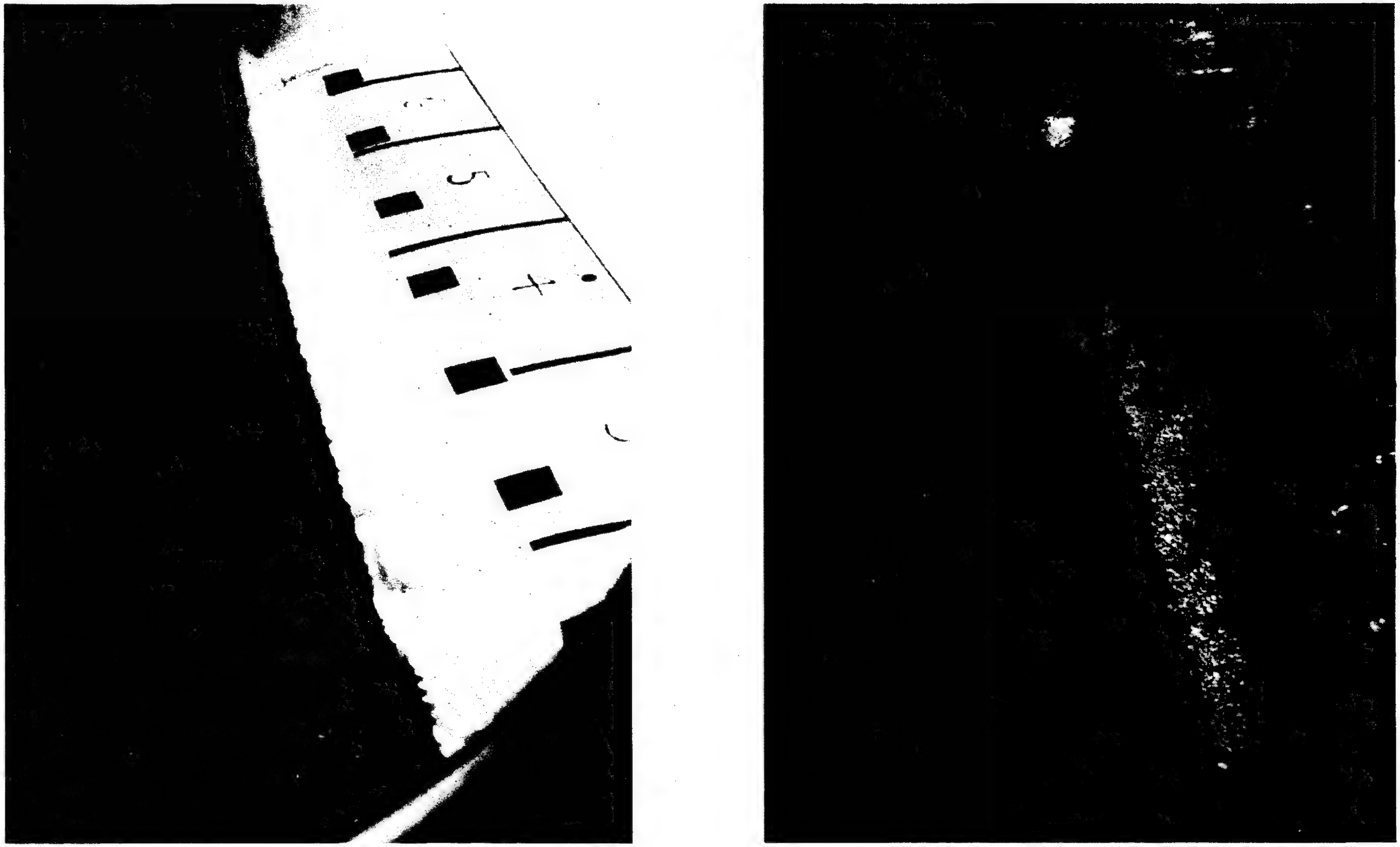


Figure 101. - BAC1-11 inflight testing. (Courtesy British Aerospace/Lucas Aerospace)

detailed investigation into its application to any particular aircraft. Rib spacing, skin thickness and other physical design parameters must all be duly acknowledged to take advantage of the EIDs. Fatigue stress is a consideration. Because of the allowable low ratio of fatigue-stress to the modulus of elasticity, aluminum alloys may not be as suitable as other higher strength materials, such as titanium-alloys. However, the specific resistivity of the alloy would be high and thereby tends to inhibit the induction of eddy currents. Consequently, a driver plate or the Lockheed approach may be necessary to overcome this problem. These considerations and the overall esoteric aspects of the EIDs merit NASA's strong support.

8. ENVIRONMENTAL CONTROL SYSTEM

The ECS power demands in the IDEA are the same as in the baseline, but in this case electric power is used to drive the turbomachinery, in lieu of engine bleed air.

In line with the baseline design criteria, the All-Electric Environmental Control System (AEECS) in the IDEA supplies 0.7 lb/min/passenger of fresh air and 0.7 lb/min/passenger of recirculated air. The pressurization schedule for the airplane calls for the maintenance of a 6000 foot pressure cabin up to altitudes of 35,000 feet and not more than an 8000 foot cabin at altitudes up to 42,000 feet. The cabin atmosphere differential pressure is limited to a maximum of 8.44 psi (as shown in figure 9). Other design conditions require an ability to maintain a comfortable cabin temperature of 75°F with outside ambients down to -59°F up to +104°F: humidity conditions for ground cooling are equivalent to 130 grains/lb. The heating and cooling Btu/h requirements under different conditions are shown in table 17.

Multiple choices for cooling are possible and they break down into a choice of motor driven air cycle, motor driven vapor cycle, or hybrid systems. Motor driven compressors (MDCs) typically provide the cabin pressurization needs in any AEECS design. The cooling requirement (on a hot day) is the major power demand on the ground while, in the high altitude cruise case, the power for pressurization is the major power demand (see figure 102). Cooling during the high altitude hot day condition with a fully loaded cabin can be met with ram air heat exchangers and without dynamic cooling. However, on hot day take-off/climb conditions up to altitudes of approximately 10,000 feet, some degree of dynamic cooling can still be required and there is the beginning of a pressurization demand. Like the baseline airplane, a three pack configuration is proposed.

8.1 Cabin Pressurization

Where tapping the engine compressors at the intermediate and high stages provided the energy for pressurization in the Baseline, a dedicated compressor system is necessary in the IDEA and Alternate IDEA. Electric motor drives are used for each of the three cabin compressors. Two types of motors are available: a permanent magnet (SmCo type), or an ac induction motor. The former, while described as a brushless dc type motor, is actually an ac (synchronous

TABLE 17. - SUMMARY OF IDEA AIRCRAFT ECS AIR FLOW AND HEATING/COOLING REQUIREMENT (1)

Operating Condition	SEA LEVEL STATIC HOT DAY	SEA LEVEL STATIC COLD DAY	15000 FT CLIMB HOT DAY	35000 FT CRUISE HOT DAY	39000 FT CRUISE HOT DAY	39000 FT DESCENT HOT DAY
Amb. Pressure - psia	14.7	14.7	8.31	3.47	2.86	2.86
Amb. Temperature - Deg. F	103	-40	46	-29	-44	-44
Amb. Humidity - gr/1bm Air	130	0	85	0	0	0
Mach	0	0	0.60	0.80	0.80	0.80
Inlet Pressure - psia (2)	14.7	14.7	10.4	5.11	4.21	4.21
Inlet Temperature - Deg. F	103	-40	82	27	9	9
Cabin Temperature - Deg. F	75	75	75	75	75	75
Cabin Heat Loads - BTU/hr						
Aircraft Systems (excluding avionics)	93100	93100	93100	86200	86200	66600
External (convection and solar)	114600	-221100	80800	-76100	-109600	-109600
Metabolic Sensible	94400	0	94400	94400	94400	94400
Metabolic Latent	53600	0	53600	53600	53600	53600
Cabin Air Flow - 1bm/min (3)						
Fresh	245	245	245	245	245	245
Recirculated	245	245	245	245	245	245
Avionics						
Heat Load - kw	16	16	16	16	16	16
Min. Air Flow - cfm	1800	1800	1800	1800	1800	1800
Max. Exit Temp. - Deg. F	100	100	100	100	100	100

Notes:
(1) Values Given Are Per Aircraft
(2) Based on 90% Ram Pressure Recovery
(3) Based on 0.7 PPM/Passenger Fresh Air And 50% Recirculated Cabin Air

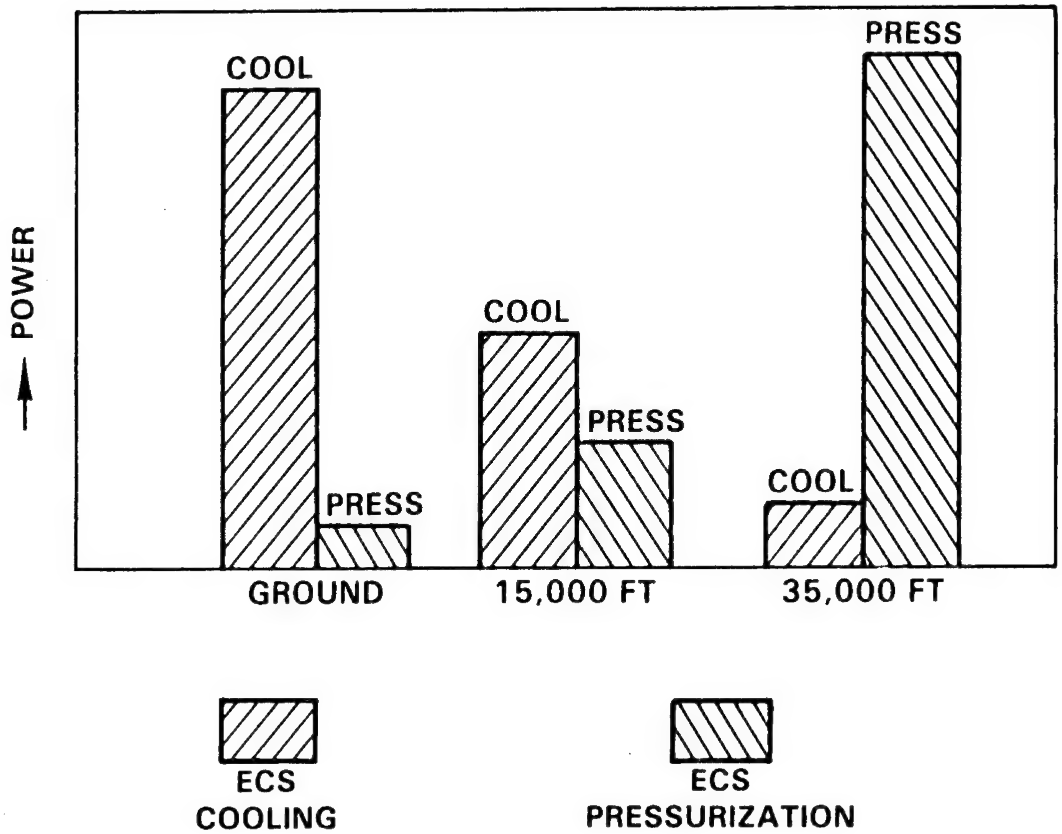


Figure 102. - IDEA ECS: cooling/pressurization comparison

type) motor that requires an electronic power assembly to synthesize the required polyphase ac power. This ac power is created by a sequential switching of power transistors, that are controlled by an electical sensor that monitors the (motor) rotor position. In the case where the ac induction motor is used as a compressor drive, the electronic power assembly is not essential, but it can still be used and the control technology would be much simpler. However, in the Lockheed selected approach to the IDEA AEECS, the induction motor driven (fresh air) compressors are directly powered from the ac system, so the weight, size, cost and thermal management problems of the electronic power supplies are avoided.

For the IDEA, a pole changing 2 speed motor and 3.2:1 pressure ratio centrifugal compressor with inlet guide vanes (IGVs) provides pressurized air for the cabin at cruise altitudes. By varying the IGV angles and motor speeds, it is possible to maintain a constant air mass flow over the operating altitude range of the IDEA.

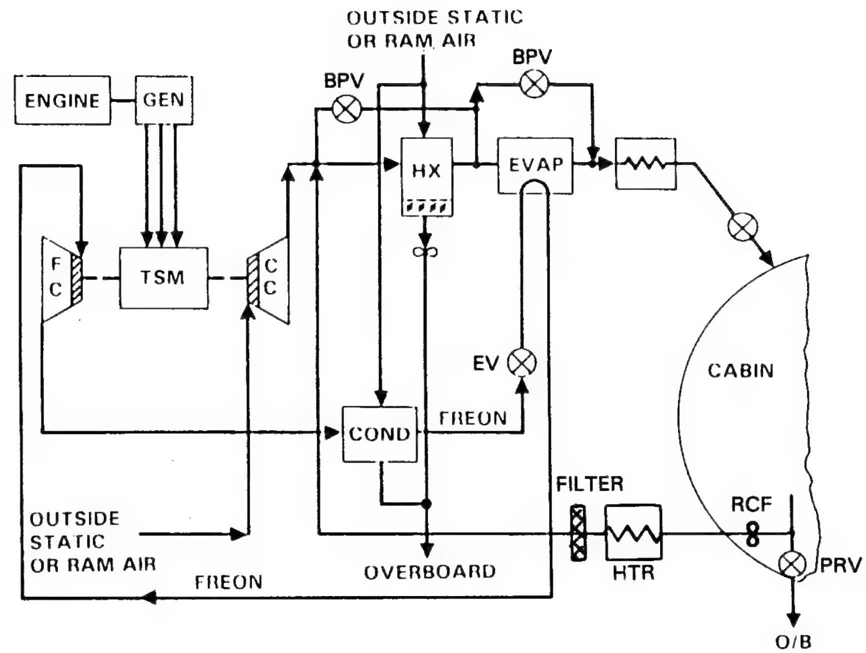
For maximum efficiency in cruise, the compressors operate with the IGVs fully open. The induction motors run at approximately 46,500 rpm (48,000 rpm minus 3% slip) with the three-phase 400 V 800 Hz power at the cruise engine speed. The compressor, which uses a titanium rotor and steel scroll housing, has a weight of approximately 75 pounds. The motor weighs approximately 100 pounds.

8.2 Cooling System

The AEECS concepts considered for cooling the cabin included motor driven air cycle, vapor cycle, air cycle/vapor cycle, hybrid cooling systems, plus variations of these systems. While a large number of configurations are possible, many have already been evaluated and those with major benefits have been identified. Air cycle concepts consisting of motor driven compressors supplying compressed air to conventional boot strap units are shown to have good performance when full advantage is taken of high pressure water separation, etc. Air cycle/vapor cycle hybrid concepts which use this more efficient cooling to meet peak hot day demands, but retain the overall reliabilty of the simple air cycle system were also evaluated. In these systems the air cycle system remains the primary mode of cooling, but vapor cycle is used to supplement the air cycle on hot humid days. Pure vapor cycle cooling is, of course, the most energy efficient cooling method and the evaporator acts also as a dehumidifier. Reliability, cost, and maintenance are some of the criticisms of the more efficient vapor cycle cooling system.

Large motors (243 hp) are required for an air cycle system. Figure 103 shows an alternative system proposed for the AEECs in the IDEA. It is a 100 percent vapor cycle system in which Freon compressors provide cooling for all hot day ground conditions and ram air heat exchangers for normal flight conditions. As shown in the table 17, the maximum cooling load is reflected by a fully loaded cabin of 350 passengers, during hot humid day conditions with a humidity equivalent to 130 grians of moisture/lb of air. Under such conditions, three 24.6 ton cooling packs are required, of which 16.8 tons is sensible heat and 7.8 tons is latent heat. Each pack is designed to supply about 200 ppm of 40°F air with a moisture content of 36 gr/lb. This provides a relative humidity of 38 percent in the cabin and it results in 100 lb of return air supply to the upstream side of the evaporators carrying some 46 gr/lb on the hot humid day. The estimated power for each freon compressor is calculated at approximately 58 hp plus 11 hp for the cabin compressor. To this must be added 20 hp for the condenser fan, 10 hp for the Hx fan and 5 hp for the recirculation fan. The Freon cooling pack power will be about 104 hp which is near the pressurization hp/pack. However, since these loads are divergent (pressurization is not required on the ground, and Freon cooling is not required at cruise flight), a proprietary approach of using a single two speed common motor drive for the Freon compressor and the cabin compressor is identified. Both the Freon and cabin compressors modulate the IGVs to control the cooling and pressurization conditions. Note that the total load reflected to the APU or external power for hot humid day cooling of the IDEA would be about 95 kW/pack or 285 kW total.

Whereas the cabin pressurization uses the two speed capability of the induction drive motor, the Freon compressor only uses the 50 percent speed condition. The point design for the vapor cycle system is therefore based on a nominal motor speed of 24,000 rpm. Motor speed changes are accomplished by changing the pole configurations on the motor as the power supply characteristics change (see below).



BPV = BYPASS VALVE
CC = CABIN COMPRESSOR
COND = CONDENSER
CV = CHECK VALVE
EV = EXPANSION VALVE
FC = FREON COMPRESSOR
GEN = GENERATOR

HTR = HEATER (ELECTRIC)
HX = HEAT EXCHANGER
PRV = PRESSURE REGULATING VALVE
RCF = RECIRCULATION FAN
TSM = TWO SPEED MOTOR

Figure 103: - IDEA ECS: electrical pressurization vapor cycle cooling.

Condition and Power Source	Power Characteristic	Motor Poles	Motor Speed (rpm)
Ground External Power	3-Ph 200 V 400 Hz	2	24,000 (nom.)
APU Power	3-Ph 400 V 800 Hz	4	24,000 (nom.)
Engine Ground Idle	3-Ph 200 V 400 Hz	2	24,000 (nom.)
Low Altitude Operation	3-Ph 400 V 800 Hz	4	24,000 (nom.)
Cruise Altitude Operation	3-Ph 400 V 800 Hz	2	48,000 (nom.) 1

1 Freon compressor off

This tabulation shows that the Freon compressor speed is compatible with the different power sources and the speed is maintained at the nominal 24,000 rpm with external power, APU power and engine driven generator power. When the airplane reaches altitudes above which dynamic cooling is not required the Freon compressor drive is isolated by opening an electromechanical clutch. As the airplane climbs to operating altitude, a point is reached where a speed change to 48,000 rpm is initiated by the microprocessor control. At this time, the IGVs on the cabin compressor close and then gradually modulate to the full-open condition, as the airplane climbs to operating cruise altitude of 35,000 feet.

As shown in figure 103, the motor drives the cabin compressor and the Freon compressor. An electromagnetic coupling is used to isolate the latter, when not required. The displacement of the cabin compressor is about 1.5 lb/sec (corrected) at altitudes up to 35,000 feet and slightly lower mass flow at altitudes up to 42,000 feet. Like the Baseline, the IDEA AEECS is designed to meet full air conditioning capability with two packs operating and an ability to maintain a 6,000 foot (pressure) cabin at altitudes up to 25,000 feet, with only one pack operating. As explained, the vapor cycle (Freon compressor) was selected as the primary ground/low altitude cooling mode but, at higher altitudes, it can be switched off to preserve its life and reduce the maintenance costs. In the ground cooling mode, the TSM operates at 50 percent speed and the IGVs are modulated to the closed position, so that the air compressor develops only enough head to overcome system pressure losses.

The heat exchanger, Hx, is used primarily to remove the heat of compression, while the evaporation provides additional cooling of the cabin inflow air. On the ground modulation of the air through the heat exchangers and condensers provides variable condenser cooling while an expansion valve (EV) modulates the Freon flow into the evaporator. Cabin pressure is modulated by the pressure regulating valve, and temperature control/mass flow control is via a μP based ECS control that furnishes the necessary logic commands to the TSM, IGVs, etc. Cabin recirculation of 50 percent is accomplished with three recirculation fans, (RCFs) that return cabin air through the filters to the front ends of the evaporators.

8.3 Advanced All-Electric Environmental Control System

Earlier in this text, it was stated that an easy rule-of-thumb relationship for a shaft powered ECS vs a bleed powered ECS was approximately 100 hp/pps. By regeneration, however, it is possible to consider a lower relationship by recirculating cabin outflow air through a cooling/power turbine mounted on the same motor shaft as the cabin compressor. In this manner the motor can be off-loaded by the amount of regenerated power.

This reduces the ECS power requirement by about one-third, resulting in a cruise ECS load on the engines of 282 hp, or 94 hp/engine. This value is used in the buildup of total secondary power loads in assessing the impact on aircraft performance (Chapter 12). The system involves technology which is proprietary to Lockheed.

8.4 Identification of Critical Technologies

8.4.1 All electric ECS. - Theoretically the technology for the emergence of an AEECS is in place today, but it is presently without any substantive support in terms of hardware implementation and integrated systems testing. Furthermore, there is the potential for the development of energy efficient AEECS approaches that would draw upon novel and innovative approaches. High altitude pressurization of large aircraft and their cooling requirements on the ground constitute the main challenges for the successful development of an energy efficient AEECS. The aerospace ECS industry has outstanding capability and expertise but they cannot independently fund this new technology when there is no new aircraft on the horizon that might take advantage of this technology.

The approaches for cooling in the AEECS could involve motor driven air cycle/bootstrap, vapor cycle, combination air cycle/vapor cycle (hybrid) and other combinations. Similarly the systems could be configured as primarily air cycle (cooling) for most normal days and supplementary vapor cycle for hot humid days. Fuel conservation will be the primary goal but the consideration of the power demand on the external power supplies is of concern as it was for engine starting. As highlighted in this report, the power capacity for cooling a fully loaded 350 passenger airplane on a hot humid day could be of the order of 282 hp. This again implies the continuous running of a high capacity APU to power the ECS on the ground with the attendant high fuel consumption and maintenance costs.

8.4.2 ECS: plans and resources for technology readiness - Estimates of the schedule and costs are shown in figure 104. Typically the resources required include aerospace ECS and electrical machinery suppliers, together with the airframe manufacturer who is responsible for integrating the system into the aircraft.

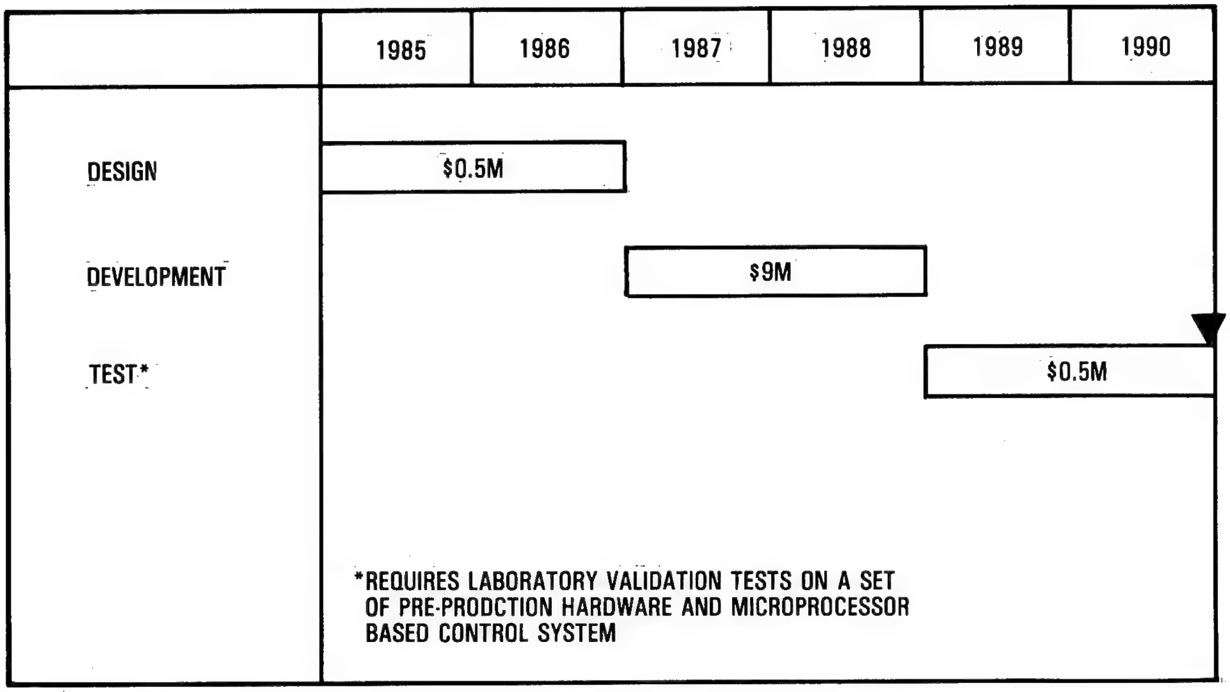


Figure 104. - All electric environmental control system.

9. FLIGHT CONTROL SYSTEM

The baseline flight control system, described in Chapter 2, represents a late 1970's flight control system (FCS) design. Its features include a three-axis, primary mechanical control system; hydromechanical actuation systems; a digital automatic FCS with a CAT III automatic landing mode; and analog augmentation and active control systems. The digital flight control system (DFCS), which includes the autopilot, flight management system, and associated avionics can be characterized as a loosely coupled group of federated multiple processors. Digital serial broadcast buses (ARINC 429) are used extensively for interprocessor communications as well as input and output to and from digital avionics.

The integration of system functions such that common sensors, I/O buses, and processors are shared is held to a minimum. Dedicated sensors and servos are used for the flight-phase-critical functions. A high degree of separation is maintained between the individual subsystems, with the subsystems partitioned according to redundancy requirements as follows:

- primary flight control system (flight-critical, mechanical system)
- basic autopilot functions and automatic landing (flight-phase-critical, dual-dual/fail-operational digital system)
- augmentation functions (fail-operational and fail-passive analog systems)

- flight management and outer-loop autopilot functions (nonflight-critical, dual/fail passive, digital system)

While the baseline architecture offers vast improvements over previous architectures, the industry has only begun to tap the potential of digital flight control systems. This potential can only be tapped by utilizing critical-path enabling technologies and methodologies which will be discussed in later sections. Integrated flight control systems technology combined with DFCS assurance methodologies will enable certificable, fault-tolerant, integrated flight control system designs while VLSI/VHSIC will enable the design implementations to meet the cost, weight, resource, and flight safety requirements. This baseline flight control system is used to define functional requirements, equipment and equipment interfaces, flight control surface configurations, and electromechanical actuation requirements for the IDEA aircraft.

The IDEA flight controls study places primary emphasis on all flight-critical, flight phase-critical, and flight essential system functions. The flight management system and other associated avionics are provided with a separate data bus which reflects a general technology improvement over the Baseline. Interconnections between this system and the flight control system is provided as needed. This separates the flight controls, propulsion controls, and power management systems from the nonflight-critical communication, navigation, flight management, performance management, cockpit management, displays and subsystem control systems. Additional architecture configurations as well as tradeoffs and analysis are considered for these nonflight-critical systems.

The electromechanical actuation system (EMAS) configuration generation, trade-offs, and analysis are documented in Chapter 6. Only the actuation interface are considered as a part of the flight controls study.

The major flight control system objective for the IDEA study is to select a certificable, full-time, flight-critical, digital flight control system which uses only electrical forms of power. This DFCS configuration is an integral part of a viable IDEA configuration. This selected configuration must demonstrate a superiority over other potential configurations with respect to selected criteria assuming potential enabling technology within the 1990 time frame. Figure 105 diagrams the overall strategy for the IDEA FCS study.

It is assumed that four independent electrical power sources with no correlated faults will be provided. Each of these independent sources have a maximum failure rate of 1 failure in 100,000 hours.

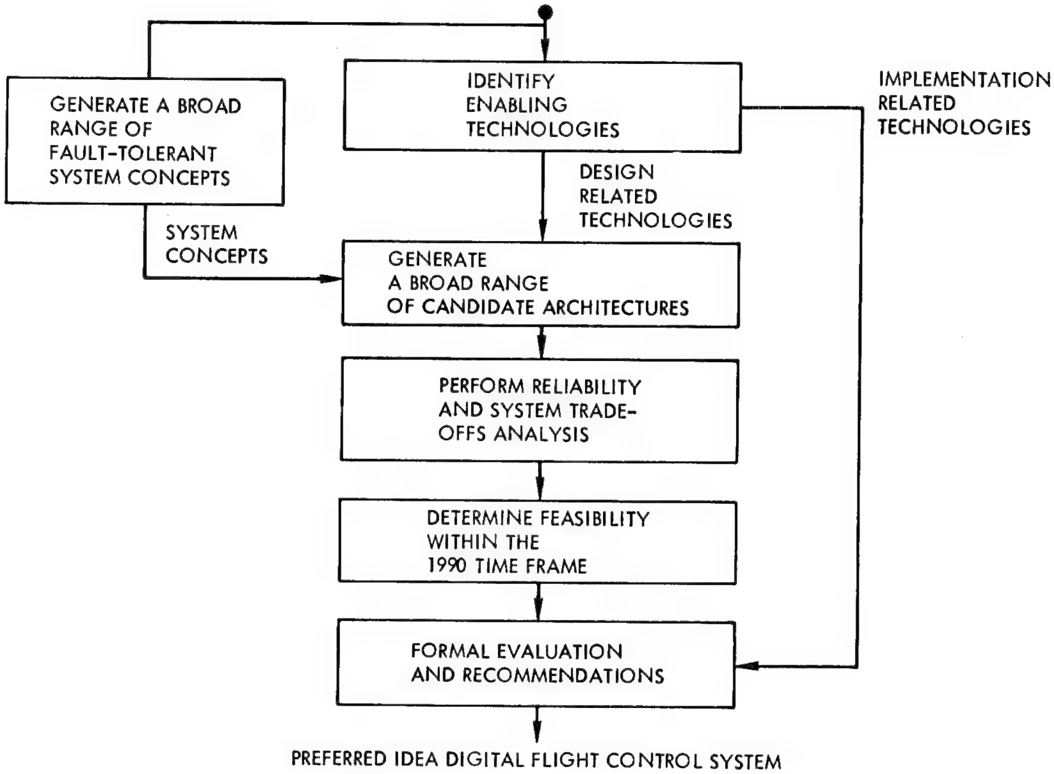


Figure 105. - IDEA flight control system study strategy.

9.1 IDEA Flight Control System Requirements

9.1.1 Design requirements. - A certificable, flight-critical DFCS that uses only electrical forms of power must be an integral part of a viable IDEA configuration. This requirement dictates that a fly-by-wire or fly-by-light primary flight control system with an electromechanical actuation system interface will be a design specific requirement. In addition, this all-electric, fly-by-wire/light technology must be ready by 1990.

9.1.2 Flight safety requirements. - Flight safety is the overriding concern for flight control system definition and validation. In order to have a certificable FCS the system configuration must comply with all applicable FAA regulations. In particular, FAA Advisory Circular (AC) 25.1309-1 defines maximum failure probabilities for flight-critical and flight-essential systems as follows:

- CRITICAL - Functions whose failures would contribute to or cause a failure condition which would prevent the continued safe flight and landing of the airplane must have a failure probability of less than or equal to 10^{-9} (extremely improbable).

- ESSENTIAL - Functions whose failure would contribute to or cause a failure condition which would significantly impact the safety of the airplane or the ability of the flight crew to

cope with adverse operating conditions must have a failure probability of less than or equal to 10^{-5} (improbable).

The primary fly-by-wire FCS and full-time pitch augmentation are classified as flight-critical systems. Since the DFCS is dependent on the electric power system, this system is also classified as flight-critical. To achieve the above reliabilities for flight critical systems a minimum design guideline is that these systems have double fail-operational levels of redundancy. The minimum failure probability must be achieved for a reference flight of 10 hours.

9.1.3 Functional requirements. - All flight control system functions implemented on the baseline aircraft and described in Section 2.5 will be a subset of the IDEA FCS functions. In addition to the baseline functions the following functions will be added:

- pitch augmentation for a relaxed static stability aircraft
- roll augmentation
- flutter supression and load limiting systems (Alternate IDEA)

A full complement of quad sensor sets allow measurement of the aircraft's state variables in both the linear and rotational axis. Table 18 lists a representative set of functions and their safety classification.

9.1.4 Memory and throughput requirements. - Computer and information transfer system resource requirements are determined by the memory requirements, computational throughput requirements, and sensor/effector I/O data requirements for all functions required and they are also dependent on the individual architecture and topology of each candidate. Tables 19 and 20 list the software/hardware functional requirements and the sensor/ effector data rate requirements respectively. Once the candidate architectures have been configured the resource requirements for each candidate can be computed.

9.2 IDEA Flight Control System Functional Description

The functional description of the IDEA FCS is almost identical to the baseline aircraft except for the following important areas.

- fly-by-wire/light primary flight controls and propulsion controls
- pitch augmentation for a relaxed static stability aircraft
- series autopilot and trim
- additional nonessential augmentation and active controls

TABLE 18. - FCS FUNCTION CRITICALITY

Axis	Function	Safety & Classification	Flight Segment(s)	Comment(s)
Pitch System	Manual Control	Critical	All	Manual Control & Stability Augmentation
	Pitch SAS	Critical	All	
	Autoland	Critical	Landing	Below Cat. III Minimums
	Pitch Trim	Essential	All	
	Stall Warning	Essential	All	
	Autopilot	Nonessential	All	
	Gust Alleviation	Nonessential	All	
Thrust	Throttle Setting Autothrottle	Critical Nonessential	All	Fly-by-Wire
Plunge	Direct Lift	Critical	Landing	Below Cat. III Minimums; Spoilers #1-4
	Load Alleviation	Nonessential	All	Automatically Deactivated via Pitch SAS Degredation; Outboard Ailerons
	Elastic Mode Suppresion	Nonessential	All	
Yaw	Manual Control	Critical	All	
	Autoland	Critical	Landing	Below Cat. III Minimums
	Yaw SAS	Essential	All	
	Rudder Trim	Nonessential	All	
Roll	Manual Control	Critical	All	Spoilers #2-6 Full Flaps Non-Essential
	Autoland	Critical	Landing	Below Cat. III Minimums
	Roll SAS	Nonessential	All	
	Aileron Trim	Nonessential	All	

*FAA AC 25.1309-1

TABLE 19. - SOFTWARE/HARDWARE REQUIREMENTS

Requirement* / Function	Throughput (kips)	Main Memory (16 Bit)	Mass Memory (16 Bit)	Loop Freq (Hz)
Primary Flight Controls (FBW)	350	30K	30K	5 to 100
Autopilot	70	15K	15K	10 to 25
Thrust Management	25	10K	10K	10
Flight Control, Display and Control (Optional)	100	40K	40K	10
Total Flight Management	505K	215K	1165K	
Radio Navigation	15	10K	10K	1 to 10
Navigation	15	30K	30K	1
Guidance	100	40K	40K	1 to 20
Traffic Control	100	15K	15K	10
Performance Management	100	10K	10K	ON DEMAND
Operations Planning	10	10K	500K	ON DEMAND
Flight Management, Display and Control	150	100K	200K	5

*Based on AIPS Requirements Study (Reference 15)

Figure 106 shows a functional description of all the manual, augmentation, and active controls used with the IDEA aircraft.

The first major difference in the two aircraft is the implementation of the primary flight controls function. They have converted from a mechanical linkage to an electrical or optical control. A side arm controller converts pilot inputs to electromagnetic impulses which are transmitted by wire or fiber optic cable to the flight computer using analog or digital coding. The computer combines this command with other input and transmits the surface position command by electromagnetic media to the EMAS using analog or digital coding. Fly-by-wire/light propulsion and engine control is another important issue that must be considered. Redundancy of the engine controllers is the responsibility of the engine manufacturer and is not a primary concern of flight controls as long as the engine reliability is not affected by the integrated, fly-by-wire/light, propulsion/flight control system. Inner loop engine control is a function dedicated to the engine controller. The autothrottle function will be implemented as in the baseline aircraft with a separate servo on the throttle quadrant. There will be quad thrust command and status signals available to the engine controllers from the flight control system.

TABLE 20. - SENSOR/EFFECTOR DATA RATE REQUIREMENTS

Requirement Sensor/Effector	Data Rate (words/sec) 16 Bit Word
Pitch Side Arm Position Sensor	50
Roll Side Arm Position Sensor	50
Pedal Position Sensor	50
Roll, Pitch, Yaw Trim Switches	5
A/P and Speed Brake Switches	1
Engine #1, 2 and 3 Throttle Quadrauts	5
Engine #1, 2 and 3 Fuel Cutoff Switches	1
Electronics Flight Instruments	525
Angle of Attack Sensor	20
Landing Gear Levers	1
Landing Gear Actuators	1
Side-Slip Sensors	20
Flap/Slats Lever	20
Radio Altimeter	50
Instrument Landing System	50
VOR	20
Digital Air Data System	50
Flight Management System	40
Outboard Aileron EMAS	170
Inboard Aileron EMAS	90
Spoiler EMAS	90
Oleo Strut Compression Switch	5
Wheel Spin Up Tach	5
Wing Tip Accelerometers	100
Flaps and Slats PDUs	30
Rate Gyros (roll, pitch, yaw)	20
Accelerometers (long. lat. vert.)	20
Inertia Reference System	620
Horizontal Stabilizer EMAS	90
Rudder EMAS	90

The second important issue is the relaxed static stability (RSS) augmentation. Operating the aircraft with a negative longitudinal static margin reduces trim drag and permits a smaller horizontal tail to be used, resulting in additional savings in weight and drag. Angle-of-attack sensors have been added to augment the pitch axis stability by commanding a positive deflection of the stabilizer for increasing (positive, pitch up) angle of attack. This function was designed as a flight-critical function since an extreme negative static margin could affect the ability of the pilot to control and land the aircraft if this function were lost.

Another important issue is that regarding parallel trim, parallel autopilots, and variable feel systems for roll, pitch, and yaw control systems. The baseline aircraft had these systems integrated into the mechanical flight

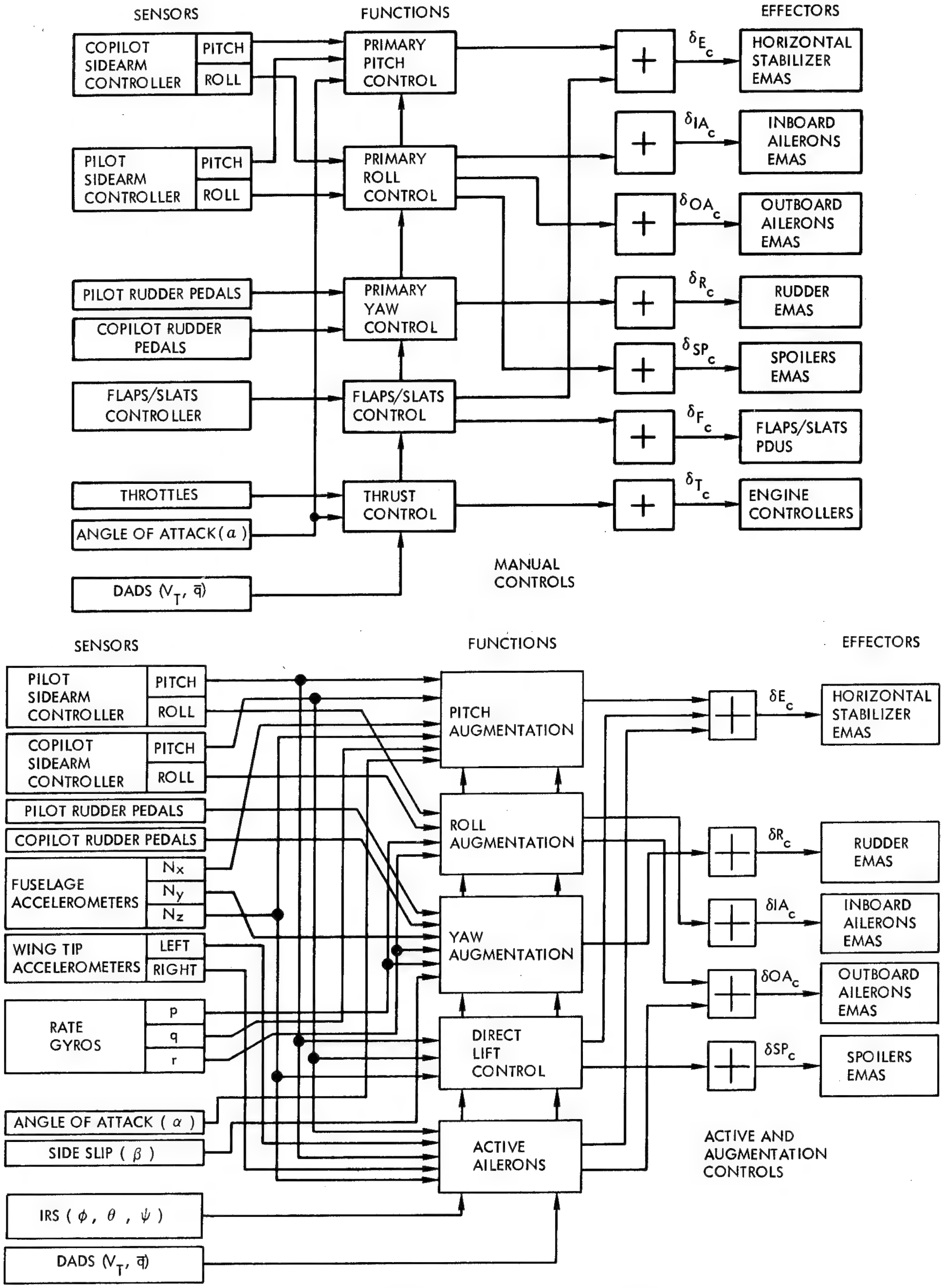


Figure 106. - IDEA FCS functional diagram.

control system. The IDEA FCS will use a side arm controller and rudder pedals for primary control of the horizontal stabilizer, ailerons, rudder, and spoilers. Controller position will command the respective surface position. All primary, augmentation, autopilot, and trim commands will be blended within the computer software and transmitted to the EMAS as one signal. This would produce only series trim and autopilot inputs without any way for the pilot to override mechanically the system as in the baseline aircraft. While a parallel trim input is not required, there may be a requirement for a variable feel system and parallel autopilot input to the side arm controller. FAR 25.1329 provides flight safety guidelines for automatic pilots. Pilot evaluation with a side stick controlled, fly-by-wire system on the Airbus Industrie A-320 should clarify flight safety and pilot preference in this area. Until the issue is resolved, the parallel trim input, variable feel input, and parallel autopilot input will remain in the system as an optional implementation. This will not affect the FCS system trades.

9.3 IDEA-based Subsystems Alternatives

Certain technologies currently under development offer performance and cost benefits which are far superior to present technologies. They also imply greater risks in achieving these desired goals by the year 1990. These payoffs are great enough to justify incorporating these technologies as a integral part of all the candidate architectures. Since they allow the design of components which would otherwise be impractical with present technology. Because of the risks, they become the critical-path in the DFCS development. The identification of critical technologies will be covered in Section 9.6. Table 21 gives a summary of the technologies which are an integral part of the IDEA candidate architectures.

9.3.1 Flight control system equipment and interfaces. - An implied requirement of all DFCS candidate architectures is that they have equipment for and interfaces to all sensors, effectors, and avionics which are required by the IDEA aircraft. The equipment and interfaces for all the DFCS architectures will remain the same except for the computers and bus hardware which will vary from candidate to candidate. Table 22 lists a comprehensive set of commercial transport FCS equipment used for IDEA. Whenever possible, the equipment will be compatible with equipment used on the baseline aircraft which is an L-1011 derivative. This will provide an accurate data base for component cost, weight, and reliability and will not affect candidate architecture trade-offs since the selection will be made on a relative basis.

The sensor and effector interfaces can be either analog or digital depending on the equipment and the architecture. The interface refers to the link between sensor/effector and the I/O bus terminal devices, which are described in Section 9.3.3.2 and figures 113, 114 and 119. Figures 107 (a) and (b) show a typical analog interface. Figure 107 (c) and (d) show typical discrete interfaces. Figure 107 (e) shows the interface for a stand-alone RTU or Mesh network node where the interface is a serial digital broadcast bus and (f) shows an embedded RTU or node where a parallel digital interface is used.

TABLE 21. - INTEGRAL ENABLING TECHNOLOGIES

Technology \ Aspect	Risk	Significance
VHSIC/VLSI	Low to Medium	Low cost, low weight, high speed components
High Temperature Electronics	High	Necessary for production of smart actuators, remote terminals, etc.
Smart Servo Electronics	Low to Medium	Extend fault detection and isolation capability
Side Arm Controllers	Low	Pilot workload reduction increase cockpit instrument visibility
Programming Languages (Ada)	Low	Embedded software design and implementation
Dissimilar Redundancy (Processors)	Low	Assurance against generic software errors
Electronic Flight Instrumentation	Low	Enhanced ADI and HSI displays Reduction in pilot workload
Fault-Tolerant Processors	Low	DFCS Processors subsystem reliability
Fault-Tolerant Power Distribution and Control	Medium	A prerequisite for all-electric fly-by-wire/light technology
Safety Assurance	High	FAA Certificability
Integral Maintenance	Medium	Adequate preflight test coverage essential to reliability analysis assumptions

9.3.2 Candidate architecture generation strategy. - Figure 108 diagrams the strategy to create an exhausive set of candidate architectures which cover the entire spectrum of possibilities. The enabling technologies and the architectural classes and options are combined iteratively to create as many realizable candidates as possible. Since the design/implementation options are so extensive their application will be addressed in the context of a set of comprehensive fault-tolerant strategies using a system level approach. A set of fault-tolerant, systems level strategies, subsystems level strategies, and I/O bus types are used.

TABLE 22. - IDEA DFCS EQUIPMENT

Type	Redundancy	Location	Interface	Flight Criticality
Computers	(Specified by Particular Architecture)			
Bus Hardware	(Specified by Particular Architecture)			
Side Arm Controller Quad Position Sensors Per Unit	2	CP	Analog	CR
Pedal Assembly Quad Position Sensors Per Unit	2	CP	Analog	CR
Flaps/Slats Controller	1	CP	Analog/Discrete	ESS
Throttle Assembly	1	CP	Digital/Analog	CR
Trim Switches	6	CP	Discrete	NCR
Speedbrake Switch	6	CP	Discrete	NCR
A/P Disconnect Switch	2	CP	Discrete	NCR
G/A Mode Switch	2	CP	Discrete	NCR
A/T Disconnect Switch	1	CP	Discrete	NCR
Individual Throttle A/T Disconnect Switches	3	CP	Discrete	NCR
Engine Fuel Shutoff Switches	3	CP	Discrete	CR
Landing Gear Controller Unit	1	CP	Discrete	CR

CP = cockpit ESS = essential
CR = critical NCR = non-critical

TABLE 22. - IDEA DFCS EQUIPMENT - Continued

Type	Redundancy	Location	Interface	Flight Criticality
Roll/Pitch/Yaw Feel System		Optional		NCR
Roll/Pitch/Yaw Parallel Trim System		Optional		NCR
Roll/Pitch/Yaw Parallel Bid		Optional		NCR
Inertial Reference Unit	3	MAB	Digital	
Instrument Landing System (ILS)	2	FAB	Digital	FPC
VOR	2	FAB	Digital	
DME	2	FAB	Digital	
Radio Altimeter (RA)	2	FAB	Digital	FPC
Digital Air Data Computer	2	FAB	Digital	
Compass	2	CP	Digital	
Electronic Flight Instruments (EFI)	2	CP	Digital	
FMCS CDU	2	CP	Digital	
FCCS CDU (Glareshield Controller)	1	CP	Digital	FPC
FACS CDU	2	CP	Digital	
Maintenance/Test Panel	1	CP	Digital	
INS/OMEGA	1	FAB	Digital	
Digital Frequency Selector	2	CP	Digital	

FAB = forward avionics bay
MAB = mid avionics bay

FPC = flight-phase critical

TABLE 22. - IDEA DFCS EQUIPMENT - Continued

Type	Redundancy	Location	Interface	Flight Criticality
IRS CDU	2	CP	Digital	
Chronometer	1	CP	Digital	
ACARS Data Link	1	CP	Digital	
FSDU/Data Link	1	CP	Digital	
FWC Display	2	CP	Digital	
Thrust Rating Panel	1	CP	Digital	
Horizontal Stabilizer Position RVDT	4	EMP	Analog	CR
Left Inboard Aileron RVDT	4	LW	Analog	CR
Right Inboard Aileron RVDT	4	RW	Analog	CR
Left Outboard Aileron RVDT	4	LW	Analog	CR
Right Outboard Aileron RVDT	4	RW	Analog	CR
Rudder RVDT	4	EMP	Analog	CR
Left Wing Spoilers #1 - #6 RVDT	24 (4/Spoiler)	LW	Analog	FPC
Right Wing Spoilers #1 - #6 RVDT	24 (4/Spoiler)	RW	Analog	FPC
Pitch Rate Gyros	4	MAB	Analog	CR
Roll Rate Gyros	4	MAB	Analog	CR
Yaw Rate Gyros	4	MAB	Analog	CR

EMP = empennage　LW = Left Wing
RW = right wing　RVDT = rotary variable displacement transformer

TABLE 22. - IDEA DFCS EQUIPMENT - Continued

Type	Redundancy	Location	Interface	Flight Criticality
Longitudinal Accelerometers	4	MAB	Analog	CR
Lateral Accelerometers	4	MAB	Analog	CR
Normal Accelerometers	4	MAB	Analog	CR
Left Wing Tip Accelerometers	2	LW	Analog	
Right Wing Tip Accelerometers	2	RW	Analog	
Dual Angle of Attack Sensors	4	CP	Analog	CR
Dual Side-Slip Sensors	4	CP	Analog	CR
Control Stick Steering Force Sensors	2	Optional		
Control Wheel Steering Force Sensors	2			
Avionics Controls Power Supplies	4	Optional		CR
Oleo Strut Compression Switch	2	LW, RW	Discrete	
Wheel Spin-Up Tack Discrete	2	LW, RW	Discrete	
Rudder Ratio Actuation System	2	EMP	Analog/Digital	CR
Horizontal Stabilizer EMAS	4	EMP	Analog/Digital	CR
Rudder EMAS	3	EMP	Analog	CR
Left Inboard Aileron EMAS	3	LW	Analog	CR

TABLE 22. - IDEA DFCS EQUIPMENT - Continued

Type	Redundancy	Location	Interface	Flight Criticality
Right Inboard Aileron EMAS	3	RW	Analog	CR
Left Outboard Aileron EMAS	2	LW	Analog	CR
Right Outboard Aileron EMAS	2	RW	Analog	CR
Left Wing Spoiler #1 - #6 EMAS	6 (1/Spoiler)	LW	Analog	FPC
Right Wing Spoiler #1 - #6 EMAS	6 (1//Spoiler)	RW	Analog	FPC
Flaps PDU Motors	2	MAB	Analog	ESS
Slats PDU Motors	2	MAB	Analog	ESS
Engine Controller Interface Eng. #1	4	LW	Analog	CR
Engine Controller Interface Eng. #2	4	EMP	Analog	CR
Engine Controller Interface Eng. #3	4	EMP	Analog	CR
MDSE Landing Gear EMAS		N	Analog	
Left Main Gear EMAS		LW	Analog	
Right Main Gear EMAS		RW	Analog	
Nose Wheel Steering		N	Analog	

N = Nose

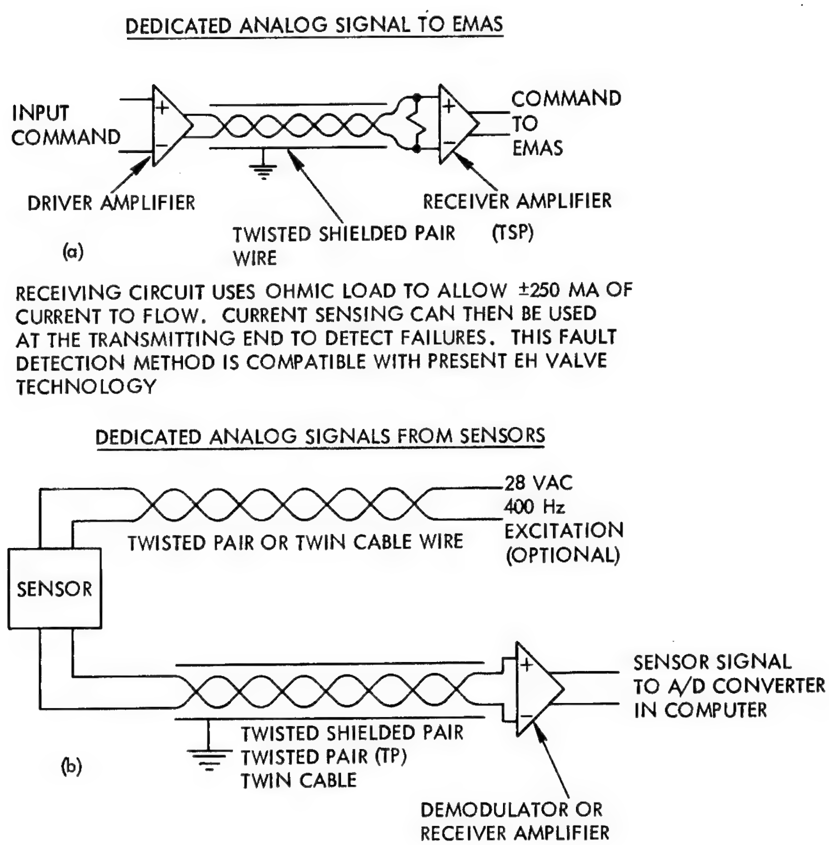


Figure 107a and b. - Sensor/effector interfaces.

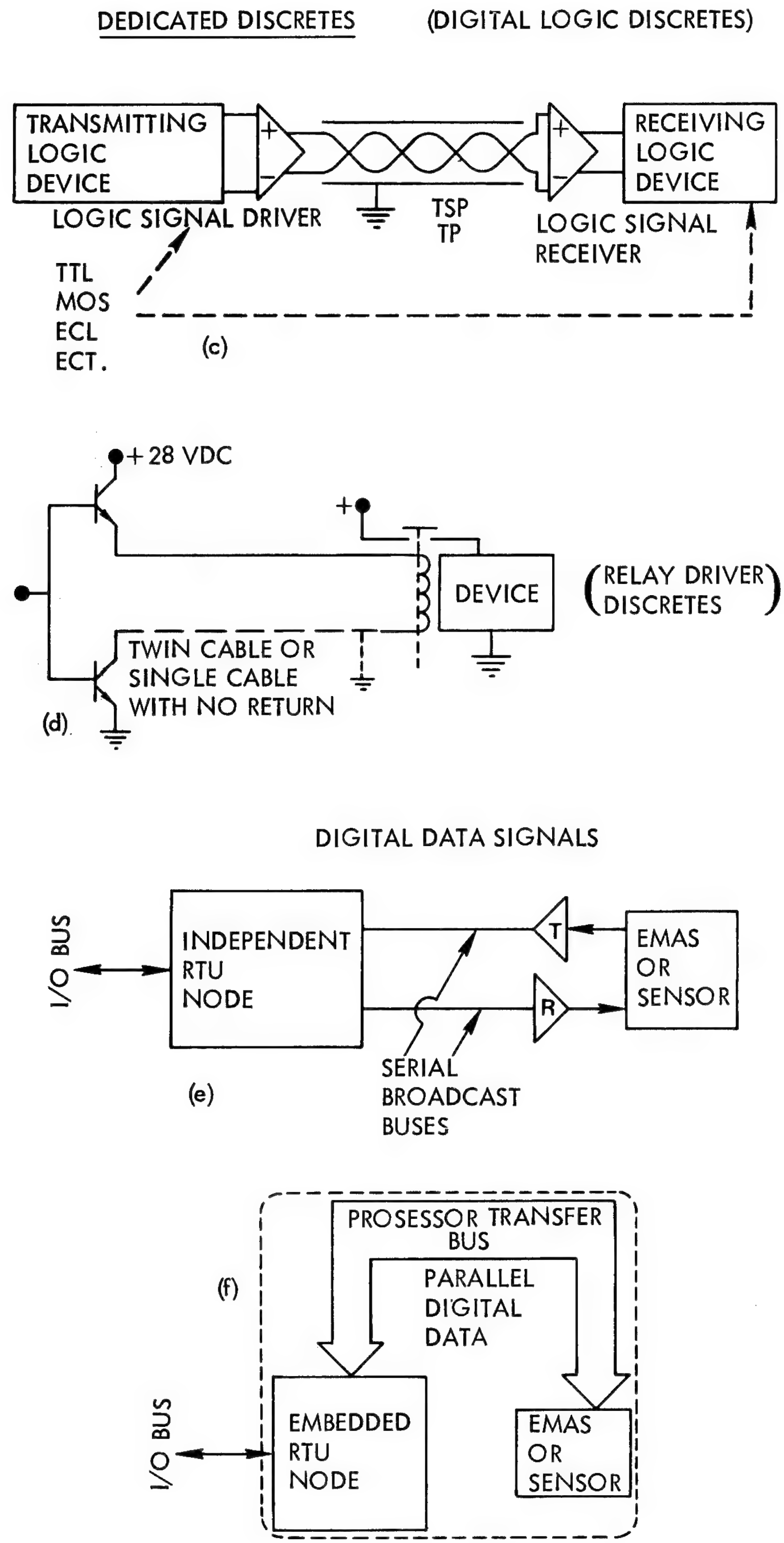


Figure 107c, d and e. - Sensor/effector interfaces.

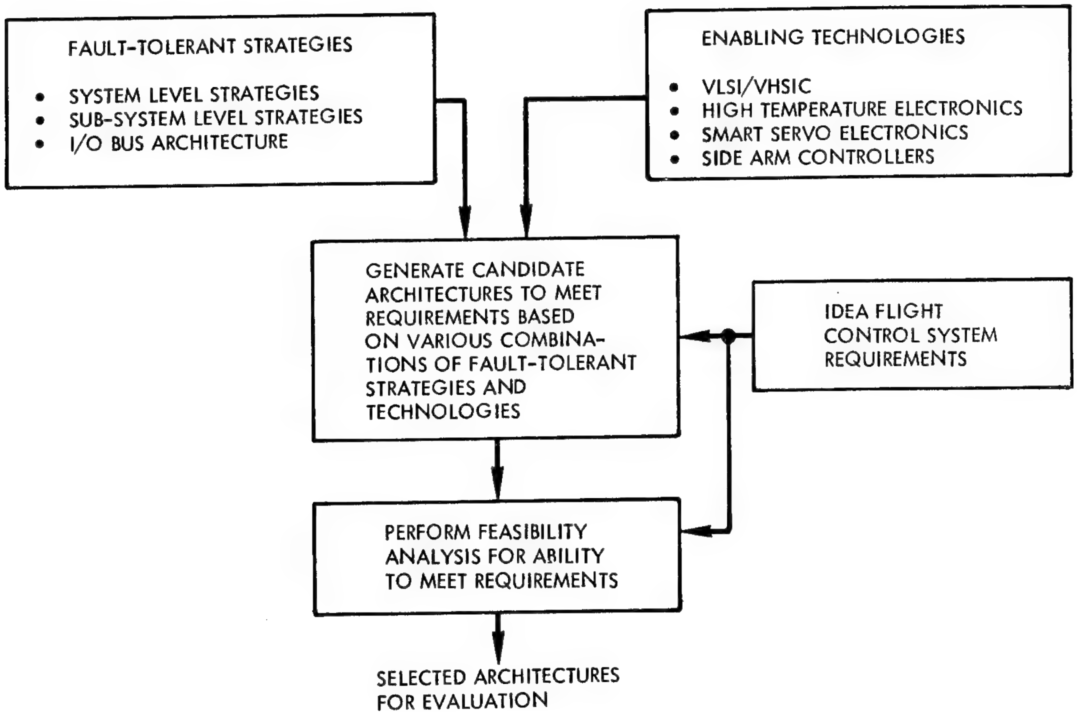


Figure 108. - Candidate architecture generation strategy.

- Systems Level Strategies

  - disjoint architecture
    The system has multiple processors where each is dedicated to a particular function and control is through independent operating systems. The processors are completely decoupled.

  - federated architecture
    The system has multiple processors where each resource is dedicated to a particular function and control is through independent operating systems. The processors are loosely coupled.

  - centralized architecture
    The functions are dedicated to a tightly coupled processor group (single resource) and control is through a central operating system.

  - distributed architecture
    The system has multiple processors where each resource is freely assignable to functions, and control is through multiple, cooperating, autonomous operating systems.

- Subsystem Level Strategies
  - parallel redundancy (triplex, quadruplex)
  - multiprocessor (FTMP, SIFT)
  - dual-monitored pairs (Honeywell's SCMP)

- I/O Bus Strategies
  - dedicated analog wire link
  - direct (dedicated serial link, ARINC 429 DITS)
  - MUX (linear multiplexed data bus such as MIL-STD-1553)
  - switched network multiplexed bus (switched network bus architecture such as the CSDL Damage/Fault-Tolerant I/O Network)
  - point-to-point networks (packet switched) star, ring, irregular, fully connected

These candidates must satisfy IDEA requirements and be compatible with the IDEA interface. Commonality among technologies and sensor/effector implementation will be maintained for all the candidates except where the particular architecture is dependent on a particular technology. Sensors, effectors, and avionics are common with the Baseline and the data communications media will be electrical (single wire, twisted pair, twisted shielded pair). This will prevent system implementations which could be applied equally to all the architectures, from entering into the trade-offs.

The implementation related technoloy trade-offs which can be evaluated and applied to the preferred IDEA candidate once the final selection has been made are as follows:

- fiber optics vs. wire

- skewed sensors or analytical redundancy

- advanced sensors

- software fault-tolerances vs. dissimilar redundant processors for assurance against generic software failures

The set of candidate architectures generated for preliminary evaluation are shown in table 23. A subset of these architectures were selected for final evaluation based on uniqueness, maturity, and level of coverage for a particular grouping. The candidates in table 23 which have an asterisk in the upper right hand corner of the first column are the selected candidate architectures. The architectures not selected were rejected because the features which made them unique from the others in its group did not give them significant enough advantages over the others.

9.3.3 <u>Selected candidate architecture descriptions</u>. - Architectures 2B, 2E, 3B, and 4A were selected for final evaulation as a representative set of potential preferred IDEA architectures. Candidate architecture 1 is not being considered as a viable candidate for IDEA because it does not represent a significant technology advancement involving high risks and high payoffs. It

TABLE 23. - IDEA CANDIDATE ARCHITECTURES

Candidate Architecture		DFCS Systems Architecture	Computer Subsystems Architecture	I/O Bus Architecture	Comments
ARINC 700 Series Fly-By-Wire	* 1	Federated	Quad, Dual-Dual Dual, Simplex	ARINC 429 Dits Dedicated Analog	Reference System
Centralized/ Dedicated	2A	Centralized	Quad	Dedicated Analog	
Centralized/ Direct	* 2B	Centralized	Quad	Dedicated Serial	128 Remote Terminals
Centralized/ Mux I	2C	Centralized	Quad	Quad Linear Multiplexed Bus	32 Remote Terminals
Centralized/ Mux II	2D	Centralized	Quad	Quad Linear Multiplexed Bus	128 Remote Terminals
Centralized/ Mux III	2E	Centralized	Quad	Dual Quad Linear Multiplexed Bus	128 Remote Terminals
Centralized/ Mesh I	3A	Centralized	Quad	Mesh Network	32 Nodes 8 BC
Centralized/ Mesh II	* 3B	Centralized	Quad	Mesh Network	32 Nodes 16 BC
Centralized/ Mesh Ring	3C	Centralized	Quad	Mesh Network	32 Nodes 16 BC
Partially Distributed/Mux	* 4A	Distributed	5 Quad Computers	Quad Linear Multiplexed Bus	Centralized Bus Control
Fully Distributed/Mux	4B	Centralized	Triplex and Dual SCMPs	Dual Linear Multiplexed Bus	Honeywell
Distributed AIPS Based	4C	Distributed	Simplex, Dual, Quad Multiprocessors	Mesh Network	Charles Stark Draper Labs

*Selected for Final Evauation (1, 2B, 2E, 3B, 4A)

is being evaluated to provide a suitable reference for comparison with the advanced architectures.

9.3.3.1 Candidate architecture 1, ARINC 700 fly-by-wire reference system: The ARINC Characteristics 700-799 define a 1980-1985 DFCS design specification for a commercial transport similar to the baseline aircraft. These characteristics specify types of functional modes, type and number of computer line

replaceable units (LRU) and their desired function, the external dimensions and the interface of the computer subsystems and avionics. The general DFCS and flight management system (FMS) architecture is specified.

ARINC 429 DITS buses are used extensively for communications between radio navigation avionics, inertial navigation system units, displays, control and display units, and computer subsystems. Analog wire signals and dedicated discretes are minimized except where required for flight safety purposes.

The DFCS and FMS can be characterized as a federated, multiple processor system made up by the following loosely coupled subsystems:

- Flight Control Computer System (FCCS) - This system contains the basic autopilot and CAT III automatic landing functions; it can be dual-dual or triplex (fail-op).

- Flight Augmentation Computer System - This system contains the flight augmentation functions and trim functions; its design specificaton is undefined and is left up to the aircraft designer. Interfaces with other systems are defined.

- Flight Management Computer System (FMCS) - This system contains the typical flight management functions plus nonessential autopilot functions; it can be a dual or simplex unit.

- Radio Navigation Systems - ILS, VOR, DME

- Inertial Reference System

The transition from present technology to fly-by wire technology can be accomplished with minimum risk by converting an ARINC 700 series DFCS to fly-by-wire technology using proven analog signals to the sensors and effectors and removing the mechanical flight control system. The column and wheel are replaced with dual side arm controllers, each with quad position sensors for primary pitch and roll control. A similar arrangement is implemented for the dual rudder pedals for yaw control. The interface to the electromechanical actuators is by analog signals and twisted, shielded pair wire. The flight augmentation computer system (FACS) is configured to provide full-time, fly-by-wire, flight-critical, primary flight controls and augmentation functions. A description of the computer architecture, I/O bus architecture, and system interconnections to sensors and effectors is shown in figures 109 and 110.

Due to the full-time, flight-critical requirement for primary flight controls the flight augmentation computer system (FACS) is a double fail-operational quadruplex unit. Signal selections are accomplished by voting signals in each computers. Fault detection, isolation, and reconfiguration (FDIR) of sensors are implemented in software. Fault detection of processors is in software with the isolation and reconfiguration implemented in hardware. The servo amplifiers are connected in such a way that they are independent of the individual processors and are driven by a voted output from the four processors. The processors are protected by wrap-arounds, self-test, and

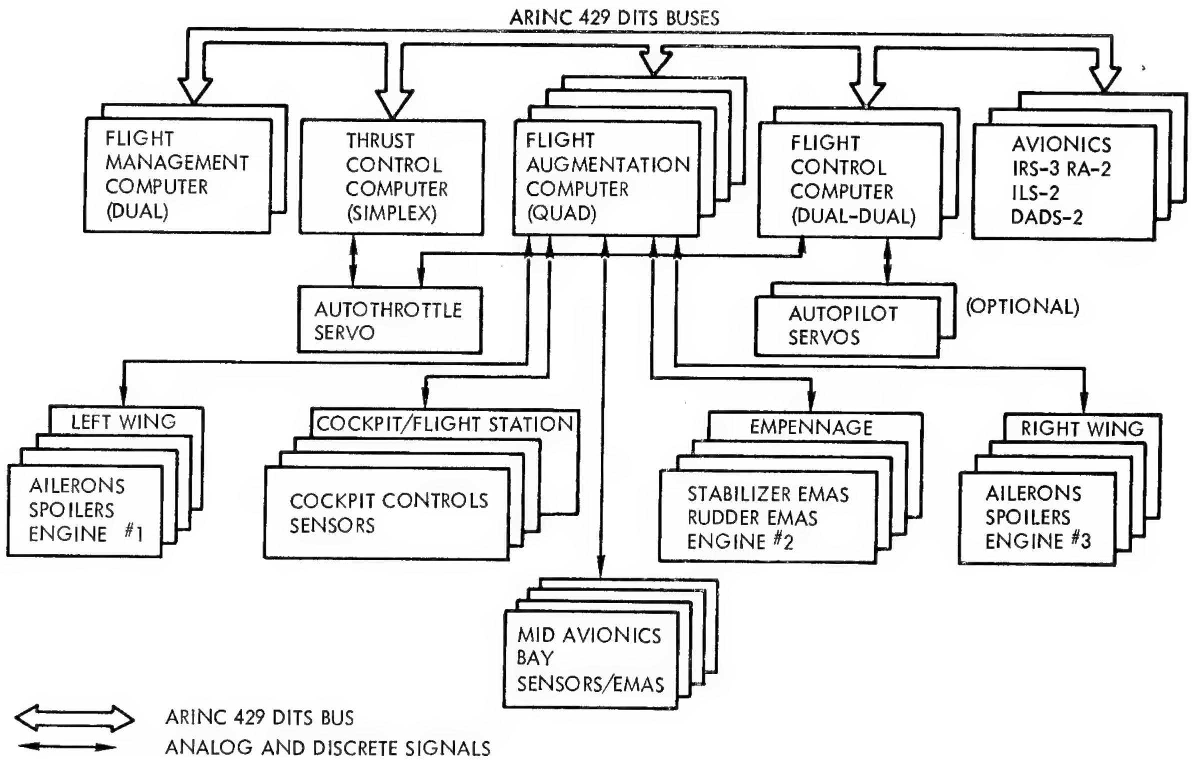


Figure 109. - Candidate architecture 1, ARINC 429 fly-by-wire reference system.

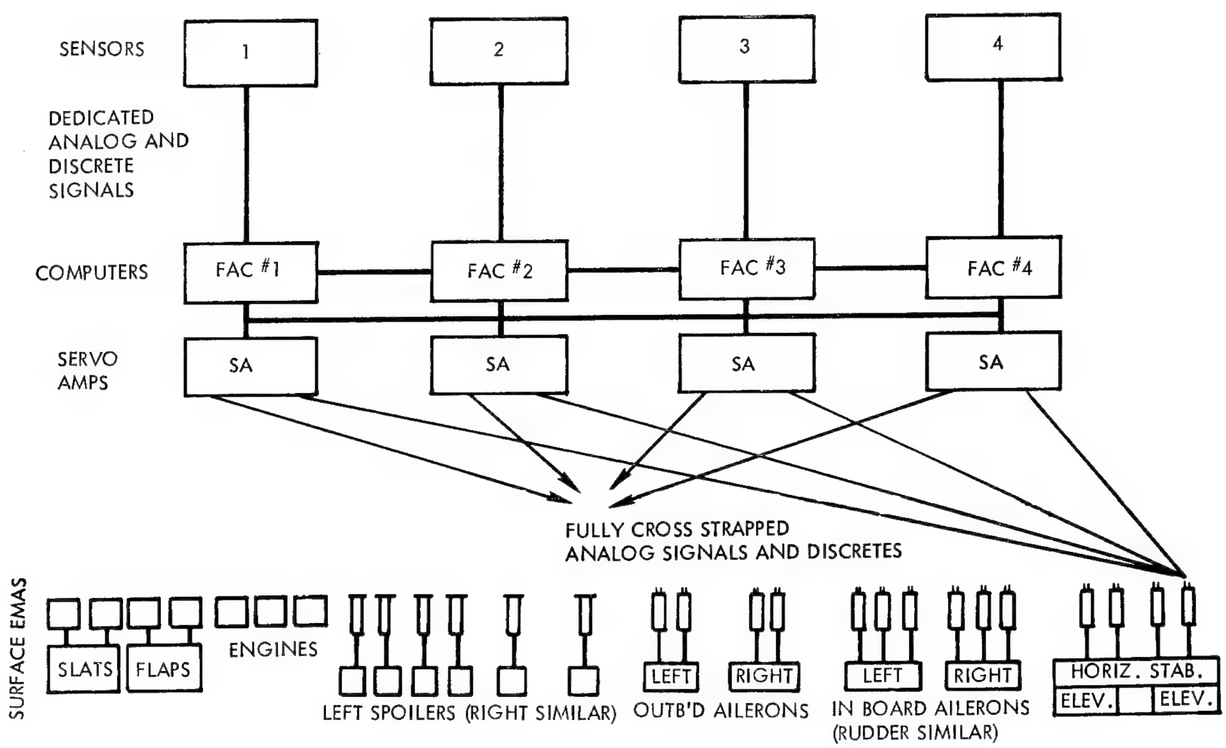


Figure 110. - Details of candidate architecture 1.

watch-dog timers. Quad analog backup computers are used to protect against a generic software failure as well as a combination of three processor failures. Interprocessor communication is by 500K bits per second broadcast buses. This allows loose frame synchronization between processors. Signal lines from the sensors to the processors are not cross strapped, but the sensor cross channel is independent from the processor. The commands and discretes to the EMAS are cross strapped.

9.3.3.2 Candidate architecture 2B, centralized/direct: This architecture has a centralized fault tolerant computer system which uses direct dedicated, serial, broadcast buses for all sensor/effector communications information transfer system (ITS). The buses are bidirectional, full duplex broadcast buses which can be either a wire or fiber optic implementation. A bus link similar to this would be constructed from two ARINC 429 DITS buses, one transmitting from the source device and one transmitting from the destination device.

The fault tolerant computer system has responsibility for the following functions:

- Bus redundancy management
  - Bus control
  - DFCS redundancy management
- DFCS maintenance/test
- Augmentation functions
- Trim functions
- Autopilot functions
- Primary and secondary manual controls (fly-by-wire)
- Propulsion controls (fly-by-wire)

Propulsion, flight controls, and bus control have been integrated into one fault tolerant computer (FTC) of unspecified design. The FTC would most likely be a processor design similar to the Charles Stark Draper Labs (CSDL) Fault Tolerant Processor (FTP) which has double fail-operational capability (references 19 and 21). Figures 111 and 112 show diagrams of this systems architecture and bus topology which uses a centralized FTC coupled to 128 remote terminal unit and transmitter/receiver unit pairs. A diagram of the RTU and T/R units are shown in figure 113. Due to the full-time, flight-critical requirement for primary flight controls the FTC is a double fail-operational, quadruplex unit. Signal selection is accomplished by voting signals in each computers. Fault detection, isolation, and reconfiguration (FDIR) of sensors are implemented in software. Fault detection of processors is in software with the isolation and reconfiguration implemented in hardware. The processor is protected by wrap-arounds, self-test, and watch dog timers. A single digital dissimilar backup computer is used to protect against a

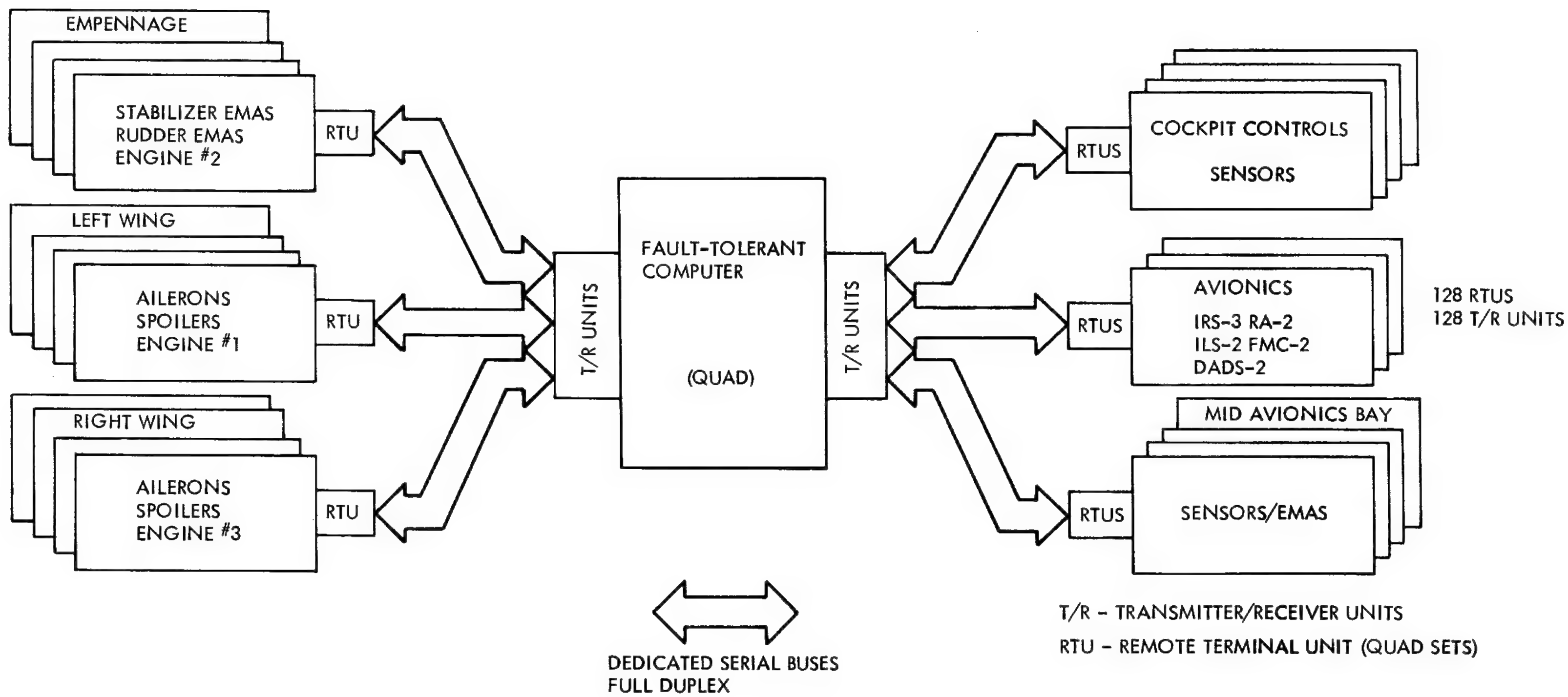


Figure 111. - Candidate architecture 2B, centralized/direct.

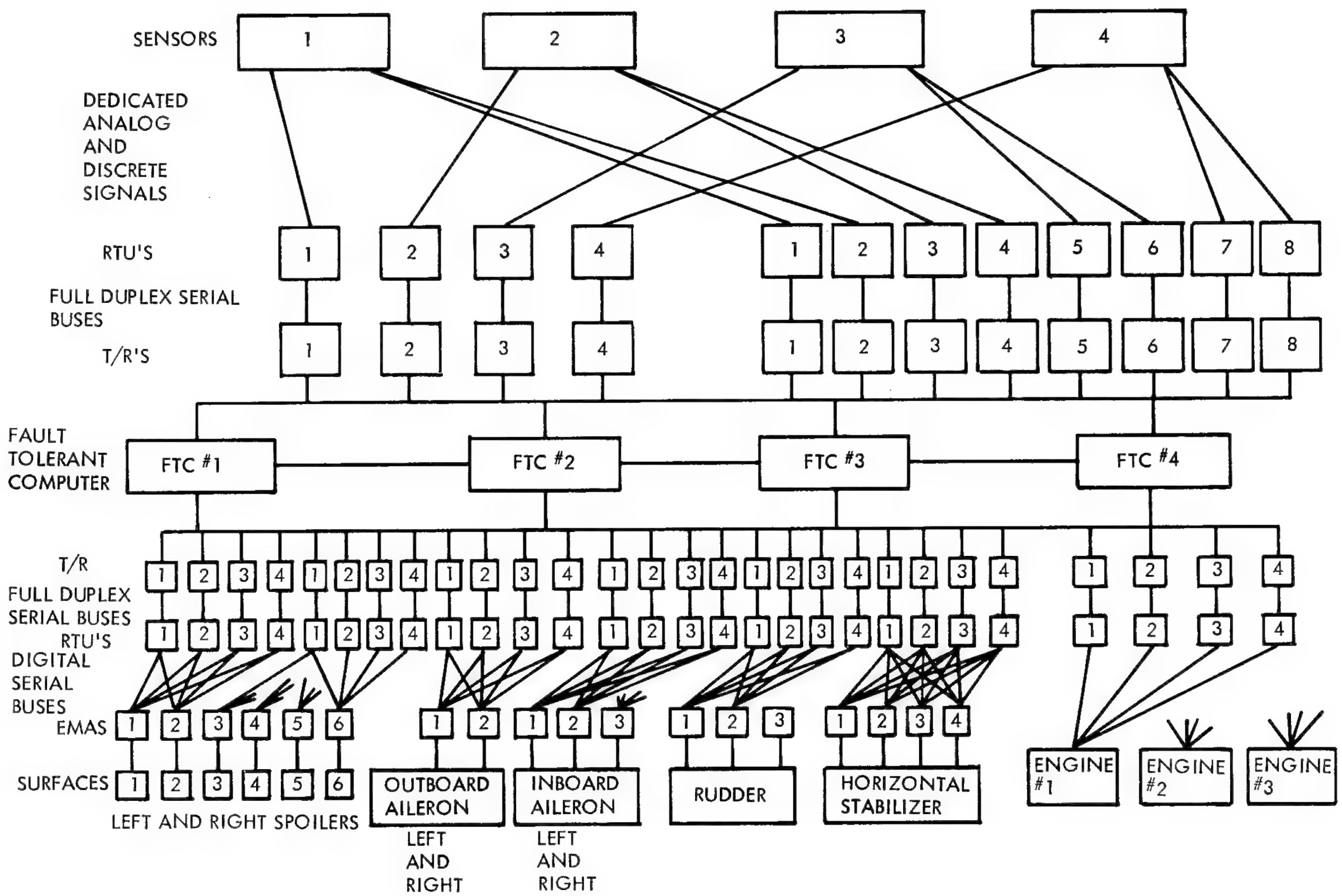


Figure 112. - Details of candidate architecture 2B.

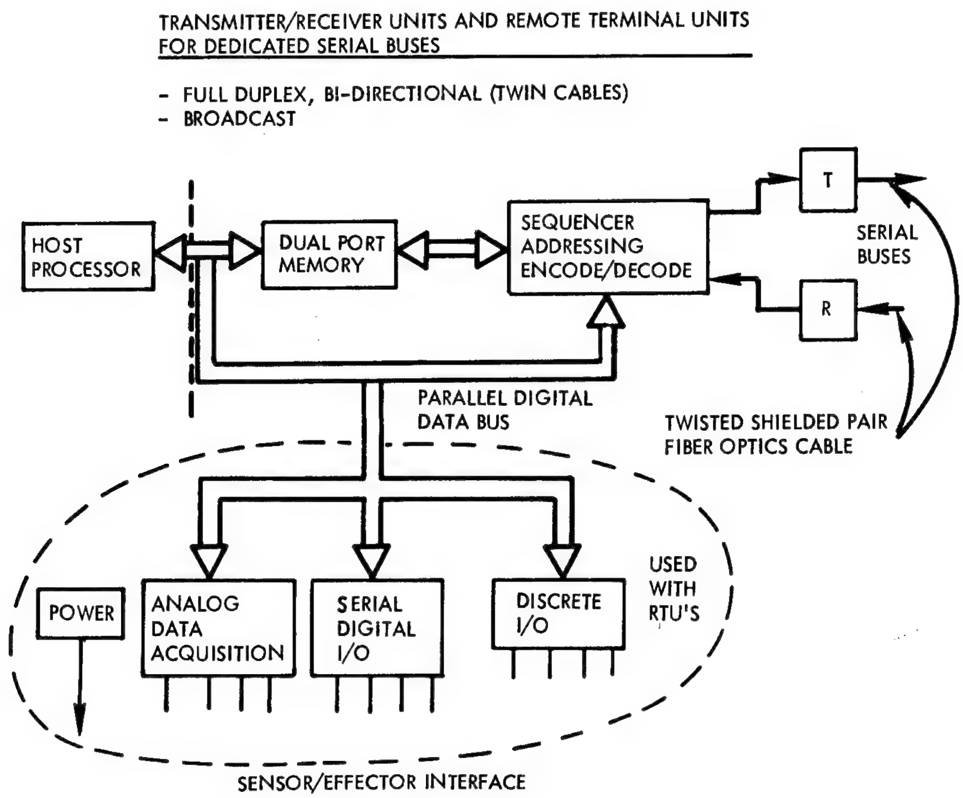


Figure 113. - Transmitter/receiver units and remote terminal units.

generic software failures as well as a combination of three processor failures. Interprocessor communication is by 500K bits per second broadcast buses. This allows loose frame synchronization between processors.

A signal transmitter/receiver unit sends and receives serial multiplexed data to one remote terminal which multiplexes/demultiplexes between one or more sensors or effectors. The basic level of redundancy is to have four T/Rs connecting directly with four RTUs which then each service one flight-critical sensor/effector group. The interface from the RTUs to the EMAS is cross strapped serial, full duplex broadcast buses. For the sensors the interface is predominately noncross strapped analog wires.

The purpose of the RTUs is to eliminate the long cross strapped wire runs that would be required if direct serial connections to each EMAS had been used without device multiplexing. Because of the large numbers of remote terminals used, it is likely that cost and weight restrictions would prevent the remote terminal units from being independent line replaceable units except where the location and interface require it. More than 90 RT units are embedded within existing LRUs (computers, EMAS controllers, avionics, etc.) and share a common power supply with the LRU in which embedded.

The T/R units are independent from the FTCs and act upon a voted set of control signals and data from the four FTCs. Thus there can be no correlated faults between FTCs and T/R units.

There are 128 remote terminals located throughout the aircraft. They are assigned to the following areas:

1. Flight Station: 28

2. Forward Avionics Bay: 32
   FTC and Bus Controller (BC) 0, 1, 2, 3

3. Mid Avionics Bay: 16

4. Right Wing: 20

5. Left Wing: 20

6. Empennage: 12

9.3.3.3 Candidate architecture 2E centralized/MUX III: This architecture has a centralized fault tolerant computer system which uses a quad, linear, multiplexed data buses for all sensor/effector communications. The buses are similar in structure and operation to a MIL-STD-1553 bus. A command response protocol and centralized bus control are used with redundant bus controllers (BC) and remote terminal units (RTU). A diagram of the RTU and BC units are shown in figure 114. Certain limitations have been relaxed with respect to bus length, stub length, and bandwidth to allow for increased capabilities due to far-term technology.

Two independent sets of quad buses are used for I/O to sensors and effectors. One quad set runs longitudinally from the flight station to the empennage while the other runs laterally from wing tip to wing tip. Four BCs are used to connect each bus to the central FTC. The purpose of the two quad bus sets is to reduce the required bus stub length and total bus length which would be required if only one set had been used. Figures 115 and 116 show diagrams of the systems architecture and bus topology.

The fault tolerant computer system and fault tolerant computer (FTC) are the same as those used with candidate architecture 2B (Section 9.3.3.2). The BCs also have the same degree of isolatoni from the FTCs as do the T/Rs in candidate 2B.

A single bus controller has centralized control over one multiplexed data bus. The command/response protocol is used to accomplish device multiplexing on each individual bus. The longitudinal buses have 22 RTUs per bus and the lateral bus has 8 RTUs per bus. The basic level of redundancy is to have four RTUs on separate MUX buses which then service one flight-critical sensor/effector group. The interface from the RTUs to the EMAS is cross strapped serial, full duplex broadcast buses. For the sensors the interface is predominately noncross strapped analog signals and twisted, shielded pair wire. Because of the large numbers of remote terminals used, it is likely that cost

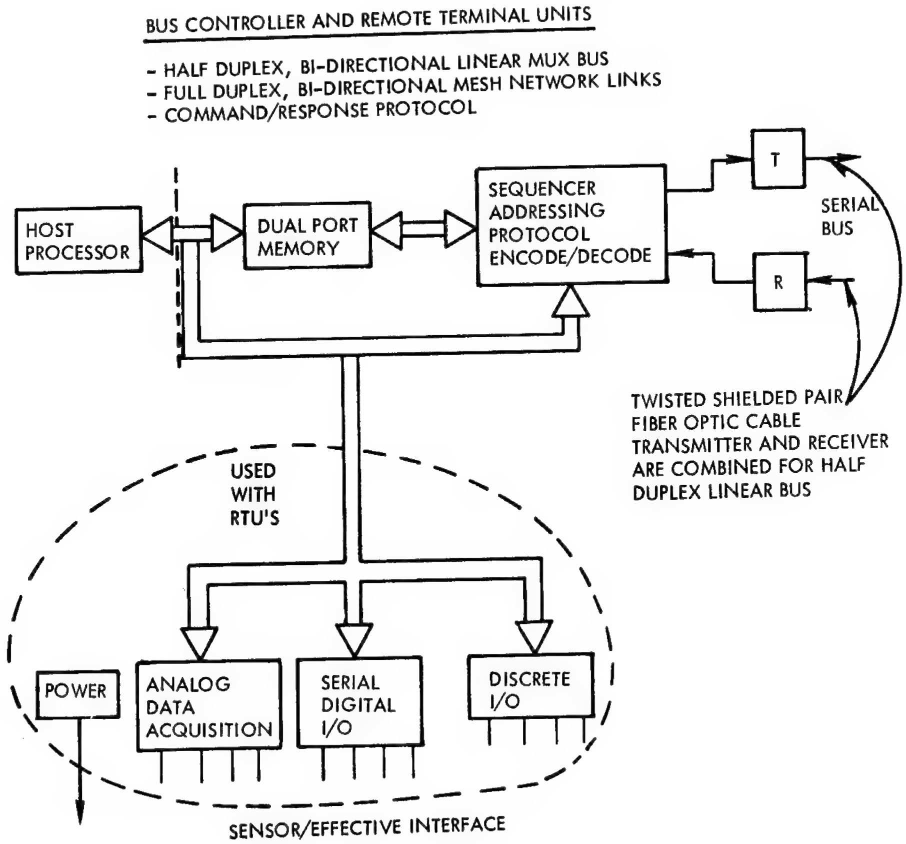


Figure 114. - Bus controller/remote terminal unit.

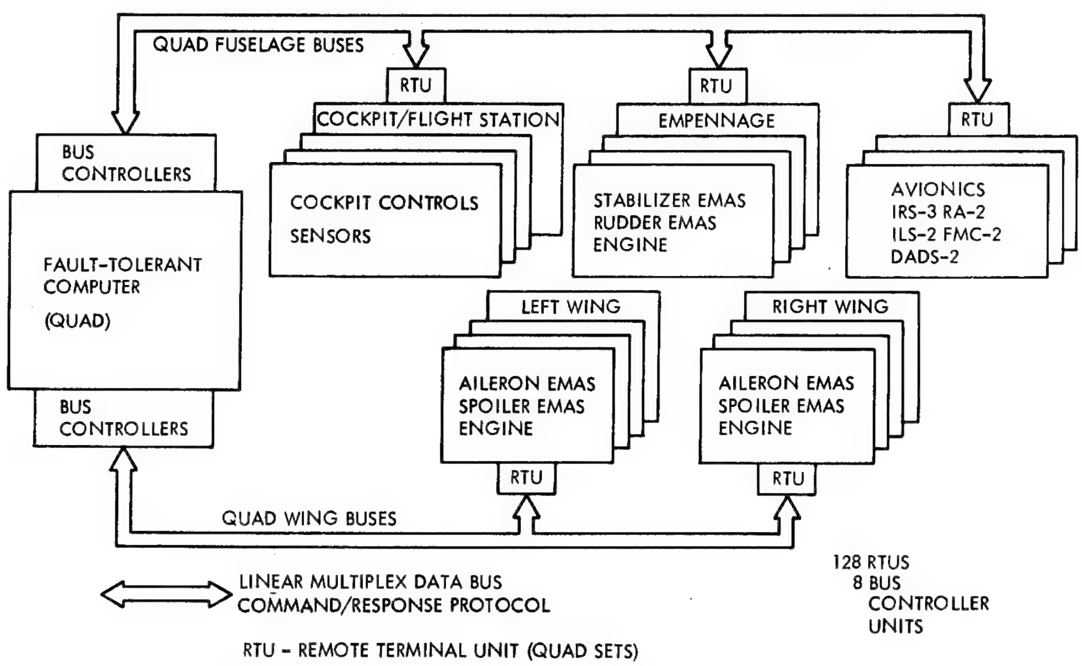


Figure 115. - Candidate architecture 2E, centralized/MUX III.

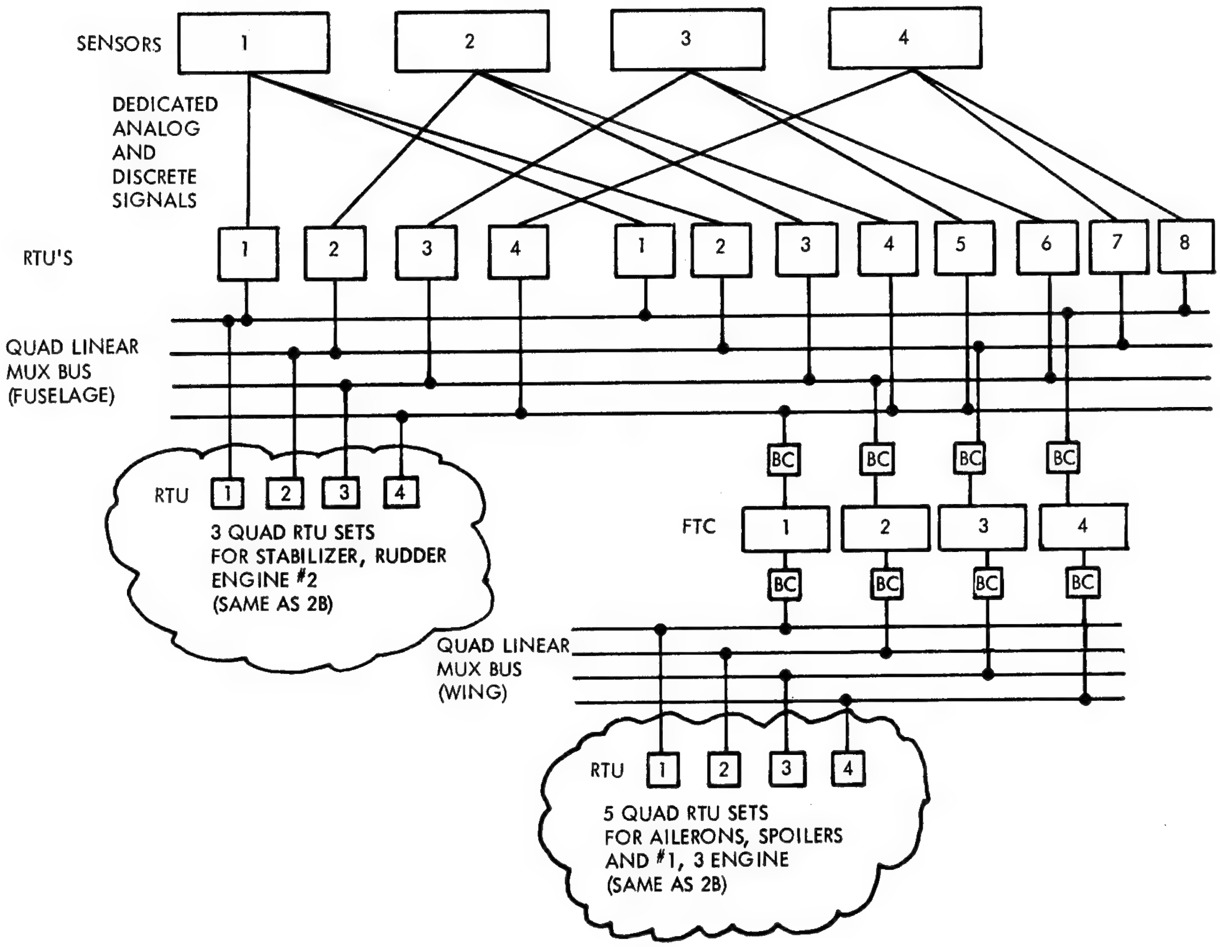


Figure 116. - Details of candidate architecture 2E.

and weight restrictions will prevent the remote terminal units from being independent line replaceable units except where the location and interface require it. More than 90 RT units are embedded within existing LRUs (computers, EMAS controllers, avionics, etc.) and share a common power supply.

There are 128 remote terminals located throughout the aircraft. They are assigned to the following areas.

1. Flight Station: 28

2. Forward Avionics Bay: 32
   FTC and Bus Contoller (BC) 0, 1, 2, 3, 4, 5, 6, 7

3. Mid Avionics Bay: 16

4. Right Wing: 20

5. Left Wing: 20

6. Empennage: 12

9.3.3.4 Candidate architecture 3B centralized/Mesh II: This architecture uses a centralized fault tolerant computer system combined with a damage/ fault-tolerant I/O mesh network using a switched network bus concept which allows path switching as well as device switching; this design was developed by the Draper Labs (Ref. 16, 17). Figures 117 and 118 show the system architecture and the bus topology. Figure 119 shows a subsystems level diagram of a mesh network node.

The network is controlled centrally by the FTC through the sixteen bus controllers which grow and reconfigure the network and control information transfers.

The nodes have three link connections and the links connecting the nodes are bidirectional, full duplex serial buses. These active nodes contain the transmitter and receiver units for the three links, a node controller, and a remote terminal which interfaces with the sensors and effectors. The node controller can configure the signal paths between the links within each node under control of the BC and FTC. Communication paths can be grown between nodes from each bus controller but none of the paths are allowed to connect. Inactive links remain to separate the active paths. Once a failure occurs, the bus can be reconfigured by deactivating active links and activating inactive links.

The fault tolerant computer system and FTC are the same as those for architectures 2B and 2E (Section 9.3.3.2). The same type of isolation is provided between the FTCs and BCs.

The bus consists of two independent strata of sixteen interconnected nodes controlled by 16 BCs and connected to one FTC. The FTC initially commands the BCs to switch a path from node to node using a command/response protocol. Once the path is switched it acts as a linear multiplexed data bus for the node RTUs on that path. Figure 118 shows a bus configuration with zero failures initially. It appears as eight linear multiplexed buses connected by inactive links. In the event of a failure the bus controller, which is continually prompting each node on its path for validity, will reconfigure the network to alleviate the fault. Either one of the two independent stratums can control the aircraft. This will allow assurance against the total collapse of one stratum. This event has to be considered highly improbable but not an impossible event since many of the nodes are interconnected and an unforeseen event could cause a fault to propagate through the entire stratum. The node redundancy design strategy is to provide a minimum of four nodes for each flight-critical sensor/effector set. In the case of the pitch control sensor/effectors, there are eight nodes. The same type of interface used for candidate architectures 2B and 2E from the RTUs to the sensor and effectors is used.

There are two types of nodes used in this architecture: independent LRU nodes and embedded nodes. Independent LRU nodes are stand-alone LRUs which

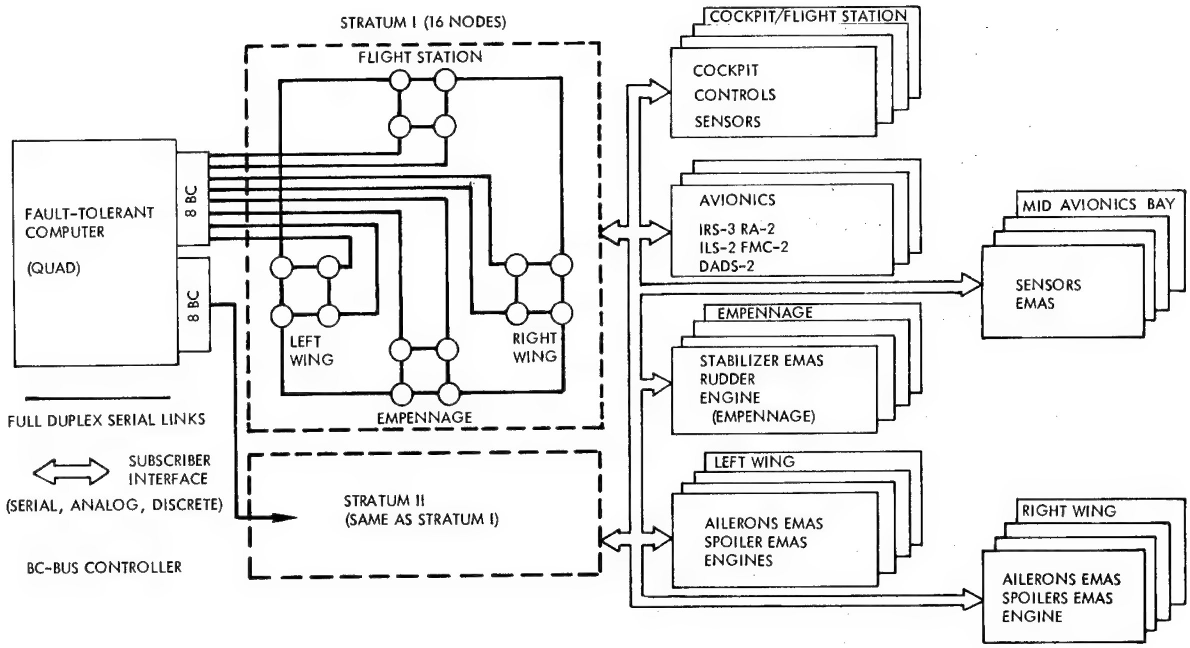


Figure 117. - Details of candidate architecture 3B.

contain a self-contained power supply, processor, and memory. The processor is used for node link control, I/O control, and self-test, etc.

The embedded node is embedded within another LRU. It utilizes the existing LRU processor, memory, and power supply. In general, embedded nodes will be contained within EMAS. The flight station, mid, and forward avionics bay nodes are LRU nodes. The wings and empennage are embedded nodes.

The 32 nodes are located throughout the aircraft. They are assigned to the following areas.

1. Flight Station: Nodes = 0, 1, 2, 3, 16, 17, 18, 19

2. Forward Avionics Bay: Nodes = 4, 15, 20, 32
   FTC and Bus Controller (BC)
   0, 1, 2, 3, 4, 5, 6, 7, 8, 9, 10, 11, 12, 13, 14, 15

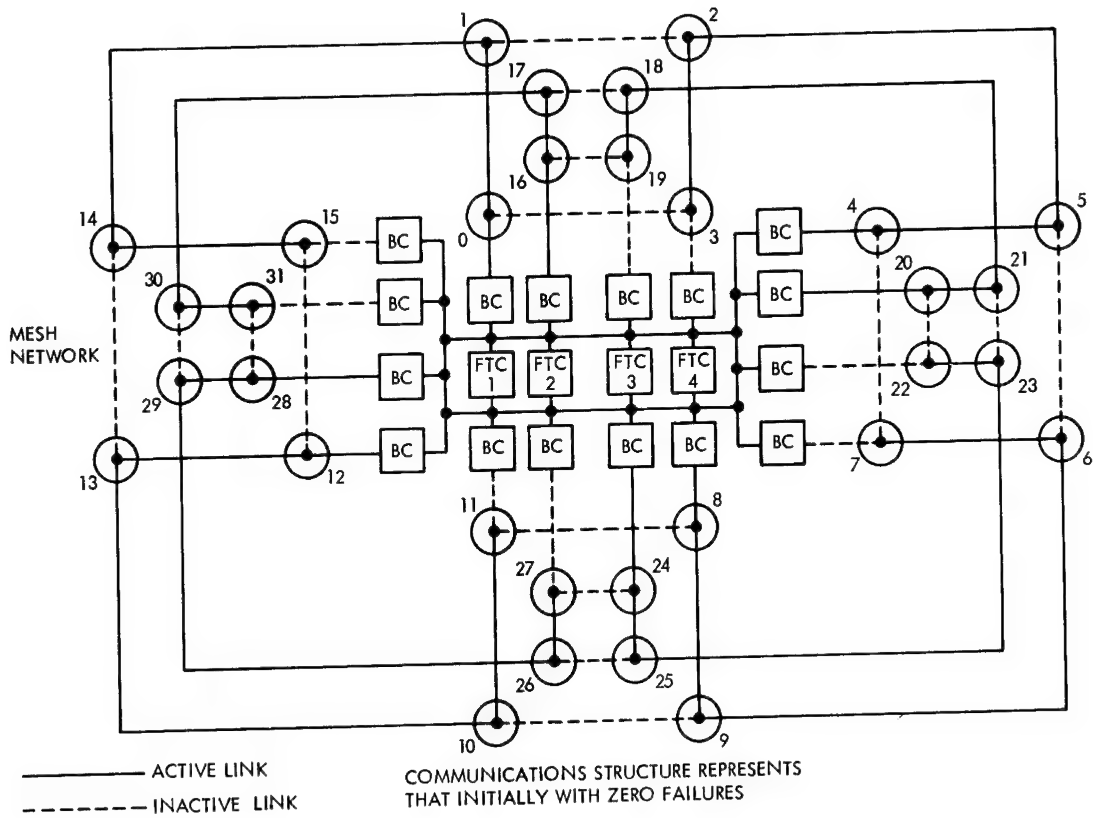
3. Mid Avionics Bay: Nodes = 7, 12, 23, 28

4. Right Wing: Nodes = 5, 6, 21, 22

5. Left Wing: Nodes = 13, 14, 29, 30

6. Empennage: Nodes = 8, 9, 10, 11, 24, 25 26, 27

9.3.3.5 Candidate architecture 4A, partially distributed/MUX: Candidate architecture 4A is a partially distributed system which uses a combination of general purpose flight computers and "dumb" RTUs distributed throughout the



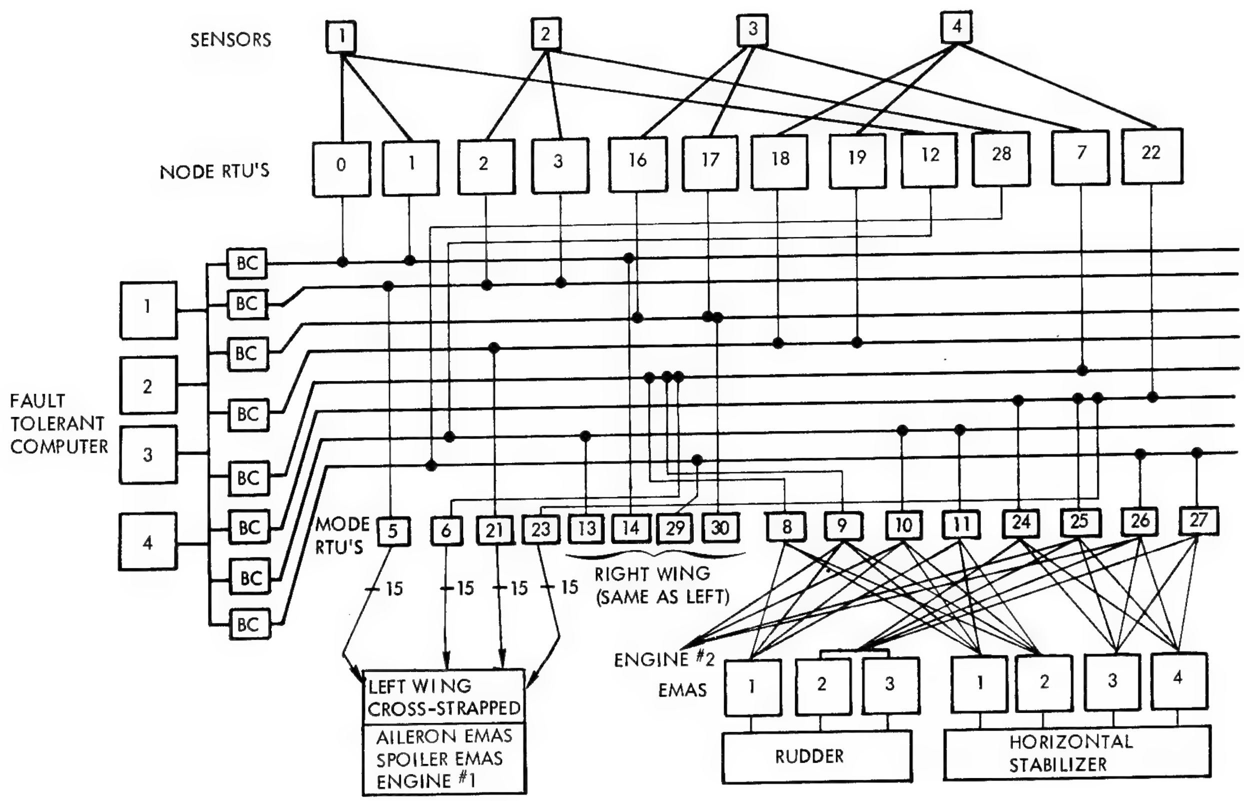


Figure 118. - Details of candidate architecture 3B.

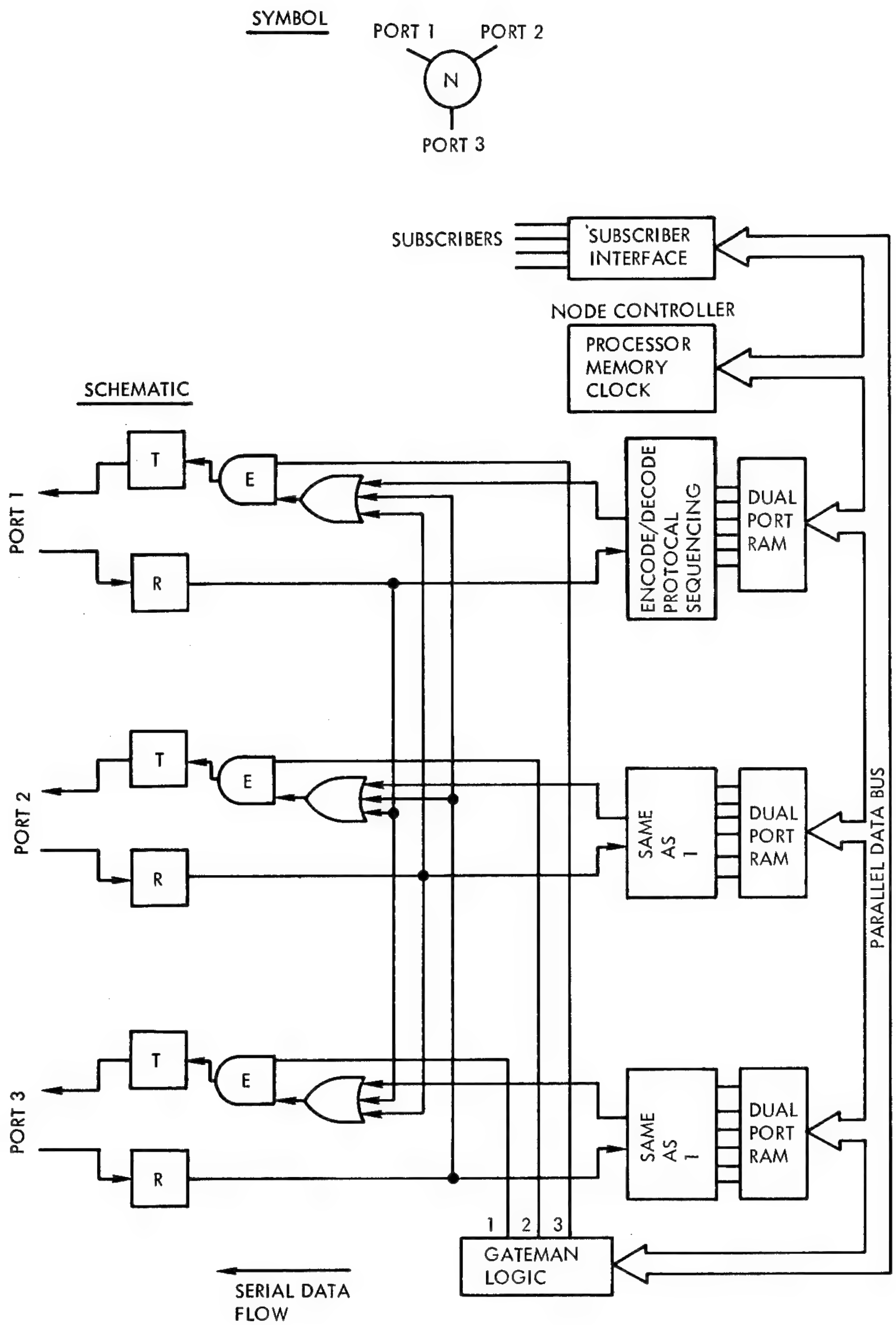


Figure 119. - Mesh network node.

aircraft. These dispersed processors and RTUs communicate through a quad, linear, multiplex bus which uses a command/response protocol similar to a MIL-STD-1553B bus. Figure 120 and 121 shows the system architecture and the bus topology for this candidate.

The "distributed" designation relates to the dispersed processing elements and to the distributed, reassignable functional processing. The fact that the system is only partially distributed is because a centralized bus controller is used instead of a fully distributed global bus control. Centralizing the bus control reduces the risks associated with the reconfiguration strategies proposed for fully-distributed systems. The disadvantages of a centralized bus controller, which relate to reliability, can be reduced by using an ultrareliable computer such as the Fault-Tolerance Multiprocessor (FTMP) or Software Implemented Fault Tolerance (SIFT). This option would result in wasted processing power and excessive hardware costs unless nonredundant processing, such as flight management, could be performed by this computer.

A second reliable bus controller option would be to use a primary bus controller with two standby backup units. The standby units will be general purpose flight computers (GPFC or FC) which normally perform flight control functions. Bus control is passed to the standby bus controller spares through a dedicated interprocessor bus which connects the three processor units. It will have serial broadcast buses and discretes which will be used to transfer bus control and redundancy management data in the event of a failure of the primary bus controller.

Much of the hardware component design has been borrowed from previous architectures. The central primary bus controller is a fault tolerant computer (FTC) having quad, parallel processors and a dissimilar hardware/ software backup processor. Four independent bus controllers are used for control: one bus controller to each bus. They are isolated from processor control in such a way that a single point of failure cannot occur on all four buses due to a single processor or bus controller failure.

The other four GPFC are quad, parallel processor units without a backup computer. They connect to the quad buses through four embedded RTUs. The RTUs in GPFC 1 and 2 are combination BC/RTU units depending on which mode they are in. The redundancy management and FDIR strategy for the GPFCs is the same as used in 2B, 2E, and 3B.

The remaining "dumb" RTUs, number 1-70, are used for sensor/effector data acquisition and communications for all devices located in remote areas of the aircraft. These are nonpressurized compartments which are exposed to extremes of heat and cold as well as vibration and shock. Because of the expected higher failure rates, the RTU redundancy in these areas will be higher. Serial commmunication buses from the RTUs to the EMAS will be cross-strapped to each actuator.

The FTC assumes bus control initially when the system is powered up. It continues to function as primary bus controller until two processors have failed within the FTC. At this point an orderly, scheduled shutdown and transfer of the bus controller from the FTC to either GPFC 1 or 2 is begun.

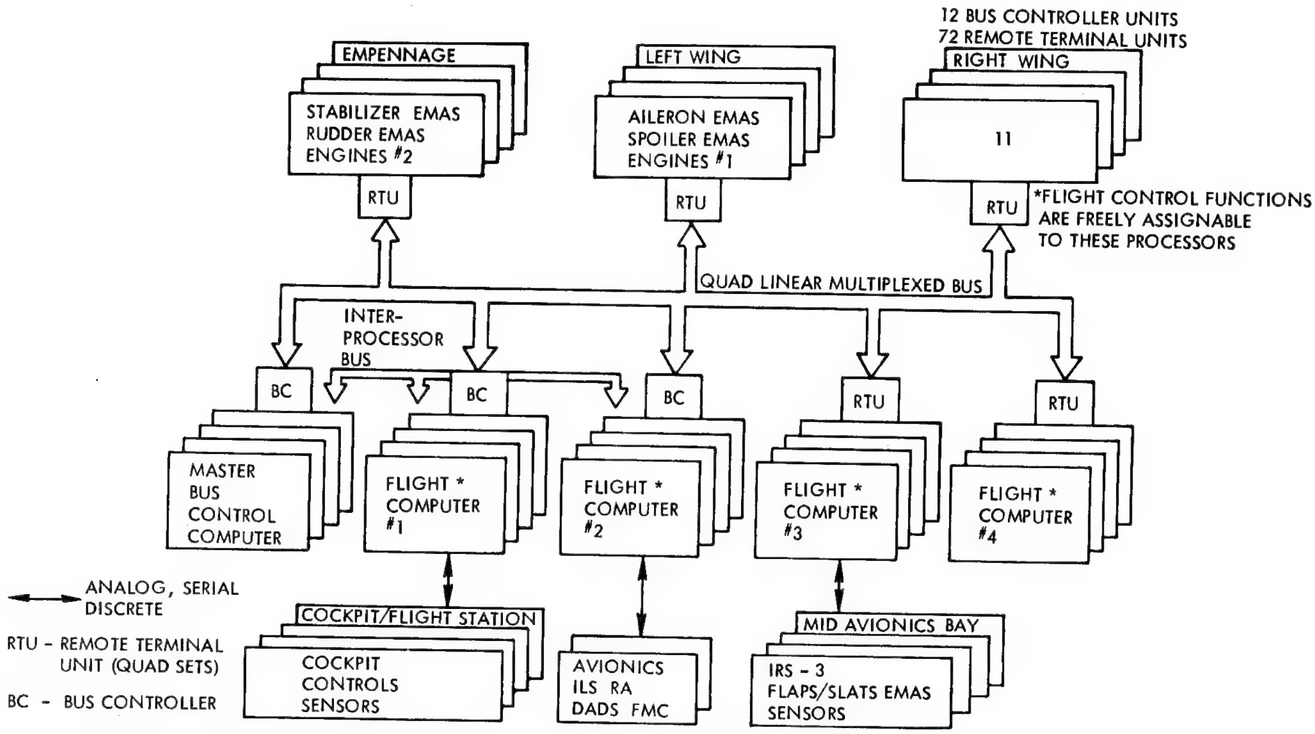


Figure 120. - Candidate architecture 4A, partially distributed/MUX.

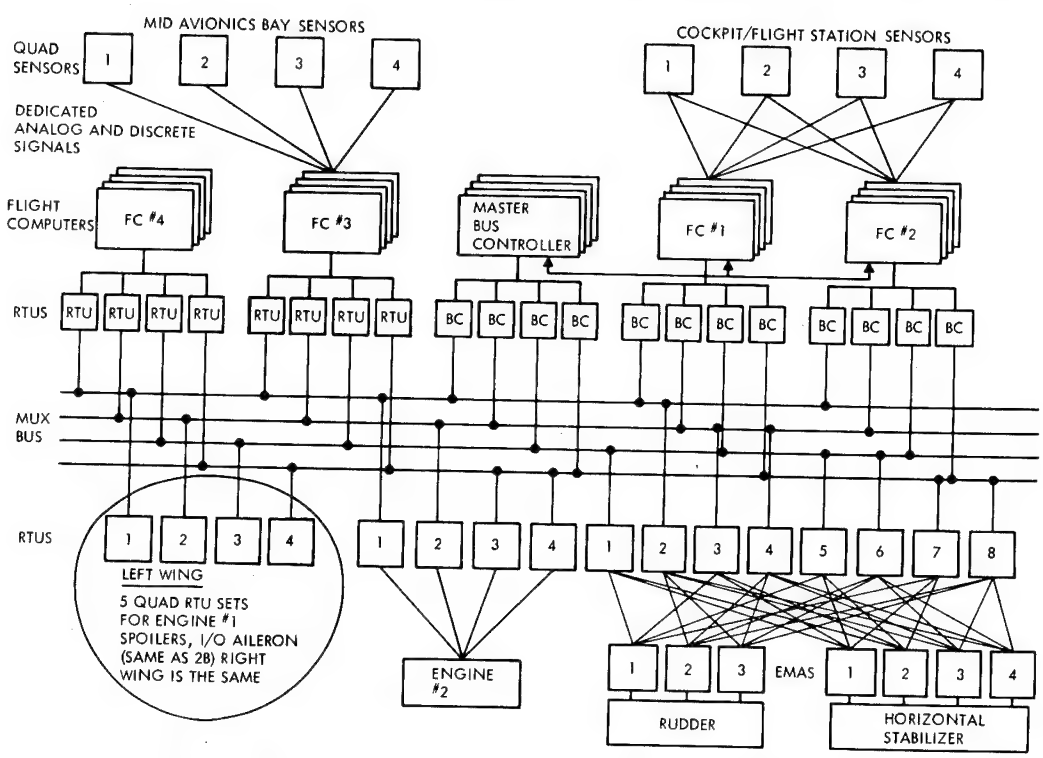


Figure 121. - Details of candidate architecture 4A.

Once completed the FTC continues to act as a backup processor for flight-critical functions if needed. After two failures have occured in the first backup GPFC bus control is transfered again to the other backup bus controller. The same strategy applies to loss of bus controllers also. This means that the system can withstand six processor failures and/or six bus controllers with no more than two of each occurring in any one FTC or GPFC.

The bus controller computer is in charge of global redundancy and systems management. It also keeps supervisory control over local and regional management which will be assigned to the GPFCs. A common global data base is kept in the bus controller and is continuously updated. This facilitates the transfer of system status data during system reconfiguration.

Idle backup processors (spares) are not used in this architecture. Each processor has a primary function, but they also have a backup function primary functional processor. In this case there are spare functional processing units which can be activated by the central controller in the event of a GPFC failure.

DFCS functions:

- bus control and global redundancy management
- primary and secondary manual flight control and augmentation
- propulsion
- automatic flight controls (autopilot and autothrottle)
- regional and local redundancy management
- sensor data acquisition and effector control

The following functions are dedicated full-time to the following processor systems:

1. GPFC 1 - sensor/effector communication in flight station, nose, and forward avionics bay.
2. GPFC 2 - sensor/effector communication in flight station, nose, and forward avionics bay.
3. GPFC 3 - sensor/effector communications in mid-avionics bay
4. RTUs 1-70 - sensor/effector communications left and right wing, empennage, mid avionics bay, forward avionics bay.

The following are reallocatable functional assignments:

1. bus control and global redundancy management

   primary processor - FTC
   secondary processor - GPFC 1 then GPFC 2

2. primary and secondary manual flight control and augmentation

   a. stabilizer and rudder

      primary processor - GPFC 4
      secondary processor - GPFC 1 then GPFC 2 then GPFC 3

   b. spoilers and ailerons

      primary processor - GPFC 3
      secondary processor - GPFC 2 then GPFC 1 then GPFC 4

3. propulsion

   a. engine 1 and 3

   b. engine 2

      primary processor - GPFC 4
      secondary processor - GPFC 1 then GPFC 2 then GPFC 3

4. automatic flight controls (autopilot and authothrottle)

      primary processor - GPFC 1
      secondary processor - GPFC 2

5. regional and local redundancy management

   These functions are concerned mainly with managing the redundancy of the sensors and effectors in a particular region of the aircraft or about a particular axis of control. They will be assigned to the processor which is actively controlling that region or axis.

A single BC has centralized control over one multiplexed data bus. The command/response protocol is used to accomplish device multiplexing on each individual bus. The basic level of redundancy is to have four RTUs on separate MUX buses which then service one flight-critical sensor/effector group. The interface from the RTUs to the EMAS is cross strapped serial, full duplex broadcast buses. For the sensors the interface is predominantly noncross-strapped analog wires. Because of the large numbers of remote terminals used, it is likely that cost and weight restrictions will prevent the remote terminal units from being independent line replaceable units except where the location and interface require it. More than 70 RT units will be embedded within existing LRUs (computers, EMAS controllers, avionics, etc.) and will share a common power supply with the LRU in which embedded.

There are 72 remote terminals located throughout the aircraft. They are assigned to the following areas.

1. Flight Station/Forward Avionics Bay: 12
   3 FTC and 12 BC

2. Mid Avionics Bay: 4

3. Aft Avionics Bay: 4

4. Right Wing: 20

5. Left Wing: 20

6. Empennage: 12

9.3.4 Flight management and avionics data buses. - Candidate Architecture 1 included the flight management system (FMS) and avionics as a part of the architectural description. The type of buses used were ARINC 429 DITS buses and the entire bus interface for the FMS and avionics was accounted for with respect to cost and weight. These same systems were included with candidates 2B, 2E, 3B, and 4A by integrating the FMS and avionics on a separate, dual MIL-STD-1553 multiplexed bus which is shown in figure 122. Separate interfaces from the FMS and avionics (RA, ILS, IRS, DADS) are provided when needed to the flight control system I/O bus. An alternate approach would be to provide redundant gateways between the two buses.

9.4 Subsystem Trades

9.4.1 Evaluation criterion. - The trades study final evaluation computes quantitative cost, weight, and flight safety reliability data for the five candidate architectures. Relative scores are assigned in these areas on the basis of the quantitative data. In addition, other qualitative assessments were made with respect to additional evaluation criteria. The complete list of evaluation criteria is as follows:

9.4.1.1 Safety related criteria:

- Flight Safety Reliability. - In addition to being a requirement, flight safety is also a criterion. Since all candidates must meet the minimum requirements of 10^{-9} failure probability for a ten hour reference flight, only the relative benefits of additional reliability margins are assessed.

- Certificability. - To design a certificable aircraft based on the IDEA requirements is one of the major objectives. A qualitative assessment of the effort to certificate the aircraft is based on all applicable FAA Advisory Circulars and Regulations.

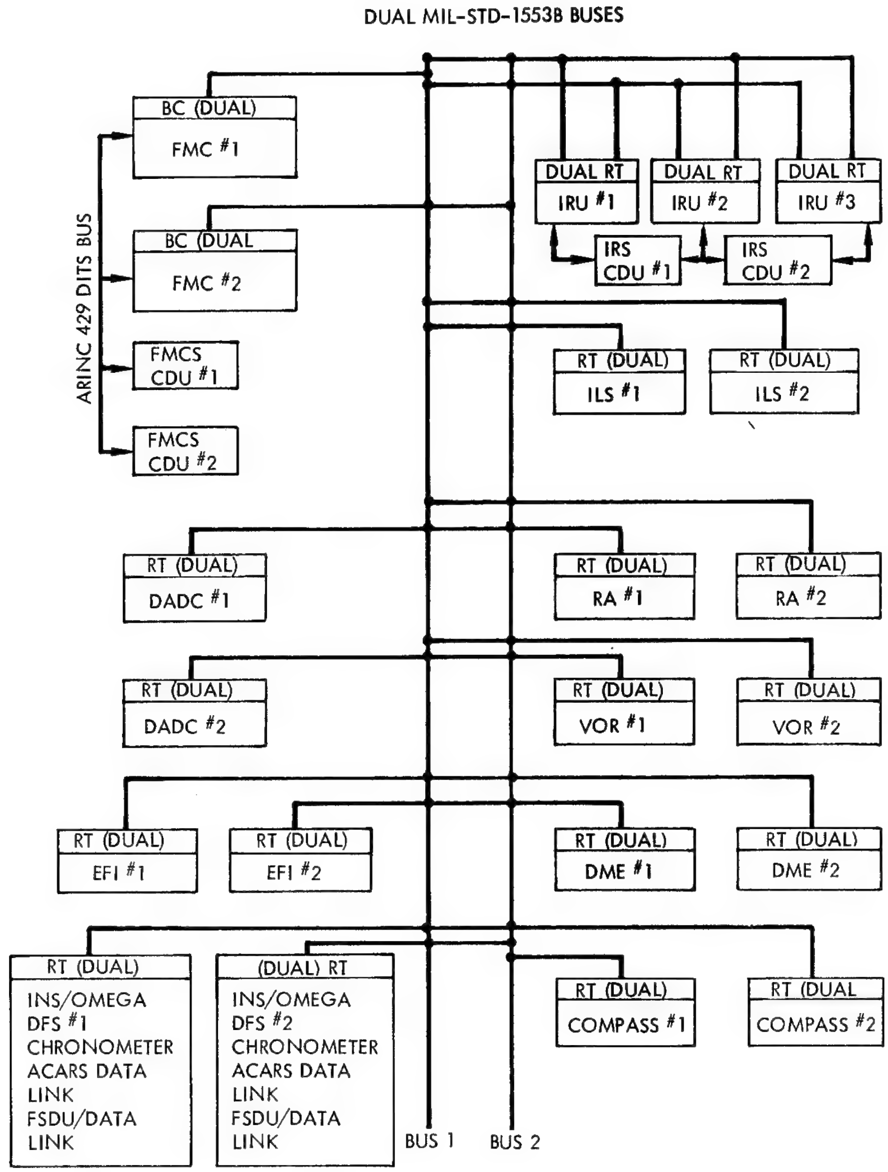


Figure 122. - Avionics bus for candidates 2B, 2E, 3B, 4A.

- Invulnerability to Environmental Hazards. - A qualitative assessment of hazards created by environmental conditions is assessed. With use of composites in an all-electric fly-by-wire aircraft, damage from lightning and strong EMI could become a safety factor. The invulnerability of each architecture to these possible hazards is also assessed.

9.4.1.2 Cost related criteria:

- Acquisition Cost. - The full-scale engineering development plus procurement cost for the FCS on 5 development aircraft and 295 production aircraft are computed.

- Weight Related Cost Penalty. - The weight of each of the five FCS is used to assess the weight related penalty.

9.4.1.3 Architectural aspects

- Technical Risk. - Technical maturity of enabling technologies and methodologies used by each of the architectures is expected by the year 1990. The possibility of failure of any of the candidate architectures from reaching maturity by the year 1990 due to complexity or immaturity of its enabling technologies are qualitatively assessed.

- Extended Development Capability. - The flexibility and future growth capability of the candidate architecture are qualitatively assessed.

- Resource Requirements. - The ability of the processor, memory, and bus technology to meet the throughput, memory, and bus capacity requirements of each candidate are assessed.

- Fiber Optic Capability. - The effects of lightning and EMI on the IDEA aircraft have not been evaluated while at the same time the decision to go completely to a fiber optic, fly-by-light technology can not be made because of the relative immaturity of fiber optics when compared to wire. A wire medium was chosen with an option to use fiber optics on the preferred system if necessary. The inability of an architecture to accept fiber optic implementation easily will imply additional cost and risk. A qualitative assessment of the ability of each candidate to accept a fiber optic implementation was assessed.

9.4.2 Trade-offs analysis results. - The previous section defined the evaluation criteria and their effect on the overall cost function which will determine the preferred IDEA DFCS architecture. This section will evaluate the performance of each candidate with respect to each of the criteria, and will evaluate the weight of each criterion and how it relates to the total cost function.

9.4.2.1 Flight safety reliability: Flight safety reliability is the probability that a flight control system will perform all of its critical functions for the duration of the mission. For this study a 10 hour flight time was used, with critical functions including manual, fly-by-wire flight controls, and stability augmentation in all three axis. Autopilot modes, with the exception of automatic landing under Category 3 conditions, are not critical. The criticality of the augmentation modes are dependent on the aircraft stability margins, so this assumption could be relaxed for some airplanes. This assumption does not affect the relative rankings of the architectures.

Failure rates, listed in table 24 for the various components, were developed from variety of sources. Failure rates for electronic devices were taken from MIL-HDBK-217D predictions made on similar units, based on a temperature-controlled, pressurized environment. High temperature circuits have been assumed to be available for use in such areas as nacelles. Gyros are expected to incorporate ring laser technology with an order of magnitude improvement in reliability.

TABLE 24. - IDEA COMPONENT FAILURE RATES

Component	Failure Rate (Failures per hour)
EMAS Signal Selector	0.75×10^{-6}
Low Voltage dc Power Supply	20×10^{-6}
Redundant Low Voltage dc Power Supply	0.016×10^{-6}
Ac Power Supply	85×10^{-6}
Redundant ac Power Supply	0.075×10^{-6}
EMAS Motor Controller	48×10^{-6}
EMAS Mechanical Parts	2.3×10^{-6}
Computer Channel	35×10^{-6}
Position Sensor	0.7×10^{-6}
Rate Gyro	29×10^{-6}
Accelerometers	74×10^{-6}
Inertial Reference Unit	100×10^{-6}
EMAS Motor	48×10^{-6}
Receiver/Transmitter Units for I/O Bus	3×10^{-6}

The electrical power supplies which convert aircraft power into low voltage dc power required by the digital circuitry are viewed as critical elements in the system. Traditionally, these power supplies have been significant contributors to the failure rate of the airplane systems using electronic equipment. The reliability of these power supplies is limited by the reliability of discrete devices such as capacitors and diodes. In the case of DFCS, it is often convenient to meet the low-voltage power needs of several components from each such supply. This practice can result in appreciable system reliability degradation if done indiscriminantly. Dual-redundant fail operational low-voltage power supplies are assumed in this analysis.

Failure rates for the EMAS are based on analysis of a unit for which detailed design is complete (Sundstrand linear, rotary C-141 aileron EMAS). Failure rates for the nodes of the mesh network configuration and the receiver/transmitter units of the bus-oriented architectures are predicted on the availability of special purpose chip sets, such as those becoming available for MIL-STD-1553B bus protocol.

Combining the component level failure rates using the Computer-Aided Redundant System Reliability Analysis (CARSRA) for the architectures other than the mesh-based candidates produced the results of table 25. Manual analysis was used for the mesh architectures. This analysis, presented in Appendix C is based on a fault tree approach. An ad hoc Fortran program was used to perform the computations.

General conclusions are as follows:

1. Candidate Architecture 1 (analog signal wires) had the highest reliability at two hours but shows the greatest degradation in reliability by ten hours (three orders of magnitude).

2. Candidate Architecture 3B (mesh network) has the highest reliability at ten hours and shows the least amount of degradation (less than one order of magnitude).

3. Candidate Architecture 2B (direct serial links) did not meet the requirements.

4. Except for 2B, double fail-operational (quadruplex) redundancy appears to be a maximum redundancy required to achieve the necessary flight safety regardless of the type of architecture.

TABLE 25. - IDEA CANDIDATE FCS FLIGHT SAFETY RELIABILITY

Unreliability	Flight Control System Failure Probability vs. Time				
Time (hours)	1	2B	2E	3B	4A
2	0.213×10^{-12}	0.961×10^{-9}	0.140×10^{-11}	0.636×10^{-11}	0.434×10^{-12}
4	0.170×10^{-10}	0.385×10^{-8}	0.112×10^{-10}	0.134×10^{-10}	0.347×10^{-11}
6	0.573×10^{-10}	0.867×10^{-8}	0.377×10^{-10}	0.219×10^{-10}	0.117×10^{-10}
8	0.136×10^{-9}	0.154×10^{-7}	0.893×10^{-10}	0.329×10^{-10}	0.277×10^{-10}
10	0.265×10^{-9}	0.241×10^{-7}	0.174×10^{-9}	0.476×10^{-10}	0.541×10^{-10}

Based on the above arguments, candidate architecture 3B has the highest reliability rating followed by 4A, 2E, and 1.

9.4.2.2 Certificability: It is apparent that architectures using near-term or present-day technology (technology that has been previously validated) have a higher degree of certificability than far-term technologies. Analog signal transmission for sensors and effectors has been used successfully for flight critical applications on fighter aircraft. Direct-coupled, serial broadcast buses such as the ARINC 429 DITS have been used for years for avionics applications. The design and operation of these serial broadcast buses are simple and robust compared to multiplexed buses where the interaction of multiple shared processors, sensors, and effectors complicates system validation.

The mesh network, while having centralized network management and control, requires complex software to grow, manage, and reconfigure the network.

Distributed architectures require complex global operating systems to manage redundancy and resource reconfiguration strategies. Communication are along a serial bus with no discrete control lines which could assure that a fault is shut down. Systems have to be designed to fail passively to assure that the bus can be controlled.

Based on these arguments, Candidate Architecture 1 has the highest rating for certificability followed by 2B, 2E, 3B, and 4A.

9.4.2.3 Invulnerability: Architectures which have a high degree of isolation between components will have a lower degree of susceptibility to lightning and EMI effect, especially when an electrical transmission media is used. A fiber optic implementation should eliminate potential problems due to the above environmental hazards; however, a fiber optic implementation could create further unforeseen problems. An architecture which can use a more mature electrical medium while at the same time minimizing environmental hazards is more desirable. Both Candidate Architectures 1 and 2B have individual, point-to point signal paths which should minimize damage in the event of a lightning strike. This assumes that the central computer has been properly isolated. The architecture which uses linear multiplexed buses has the highest susceptibility due to the length of the buses and the multiplicity of connected equipment on each bus. The mesh network has by far the lowest susceptibility due to the interconnected nodes which are electrically isolated at each node; damage would not propagate to the vital centralized computer system. Based on the above argument candidate architecture 3B has the highest invulnerability rating followed by 1, 2B, 4A, 2E.

9.4.2.4 Aquisition cost and weight related cost: Aquisition cost were computed for the computer hardware, software, bus hardware, and wire. Sensors, effectors, and avionics were common to all candidates and were not included in the selection process although they were included in the cost input for total life cycle cost to ASSET for the preferred architecture. DFCS computer cost were computed based on VLSI/VHSIC technology being developed by the Honeywell Corporation which is participating in the government's VLSI/

VHSIC development program. Reference 18 documents Honeywell's cost estimate for VLSI/VHSIC DFCS integrated circuit components used on a far-term advanced tactical fighter. The following methodology was used to compute cost and weights for the IDEA candidate architectures:

1. DFCS subsystems were configured based on a high level design scheme.

2. Chip sets were proposed to implement the subsystem based on VLSI/VHSIC technology using the same approximate level of circuit density per chip.

3. Total chip counts to implement each subsystem, and subsystem cost and weight were computed based on the chip's full-scale engineering development cost (nonrecurring cost (NRC)) and production cost (recurring cost, (RC)).

4. Total system cost and weight were computed from the subsystem data and the number of required subsystems.

5. Wire length was computed and its cost and weight was added to the DFCS cost and weight.

6. The software cost was computed using a $/(line of code) cost factor. The code length was based on the hardware and software requirements.

The total cost for each DFCS architecture is shown in table 26. Based on this quantitative assessment the candidate with the lowest cost is 3B followed by 2E, 4A, 1, 2B.

DFCS weight has the least impact on total system cost. The weight assessment is shown in table 27. The weight saving in going from 2E (lowest weight) to 1 (highest weight) is 840 pounds. Based on this quantitative assessment the candidate with the lowest weight related cost is 2E followed by 4A, 3B, 2B, and 1.

9.4.2.5 Technical risk: In general, the risks increase as the system designs incorporate far-term technologies and untested designs and concepts. The same argument used for the level of certificability holds for level of risk. The Candidate Architecture with the lowest level of risk would be 1 followed by 2B, 2E, 3B, and 4A.

9.4.2.6 Extended development capability: Architectures such as 1 has a low flexibility and future growth capability because the dedicated link can only service one sensor or effector. The addition of new devices will require additional wires to be run from the device to the main computer. The dedicated, serial ARINC 429 DITS buses also require additional buses to be added

TABLE 26. - IDEA DFCS COSTS

Itemized Cost Candidate Architecture	Wire ($K)		Computers ($K)		Software ($K)		Bus Hardware ($K)		Totals ($M)
	NRC	RC	NRC	RC	NRC	RC	NRC	RC	300 Aircraft
1	4095	344	7291	82	11520	0	0	0	151
2B	1958	164	4409	63	11520	0	19462	289	192
2E	755	63	2268	140	15520	0	19699	1285	151
3B	1405	118	2555	43	11640	0	9248	142	116
4A	1840	170	9372	143	11520	0	15390	171	152

NRC = Non-recurring cost
RC = Recurring Costs

TABLE 27. - IDEA/WEIGHT

Itemized Weight Candidate Architecture	Wire		Computers (lb)	Bus Hardware (lb)	Misc. (lb)	Total (lb)
	Length (ft)	Weight (lb)				
1	88671	1056	336	0	11	1403
2B	24219	507	35	165	37	744
2E	10049	216	20	165	37	438
3B	17710	363	66	66	37	532
4A	11030	216	75	115	37	443

for each additional avionics device. Hardware changes will be required within both the main computer and the added devices. Architecture 2B is more flexible than 1 because its remote terminals and serial buses allow additional devices to be added as long as additional RTUs are not required. The linear MUX bus architectures (2E and 4A) have the greatest capability because the linear bus allows additional devices to be added anywhere on the bus without

major hardware changes to the centralized computer. Next to the MUX bus the Mesh network bus has the best flexibility and future growth capability. Additional nodes can be added locally and linked into adjacent nodes. The architecture with the highest flexibility and extended development capability is 2E and 4A followed by 3B, 2B, and 1.

9.4.2.7 Fiber optics compatibility: Architectures which are compatible with fiber optics media have a higher risk associated with using fiber optics instead of wire because of the immaturity of fiber optics buses for flight controls applications. Some parts of the flight control system are more exposed to the EMI/Lightning environment and might require fiber optics data buses, while other parts could use wire.

Candidate architecture 1 has the lowest compatibility with fiber optic because the fiber optic transmitting devices operate best in a pulse mode rather than a digital signal. A pulse width modulated analog optical system could be designed but it would have little advantages over a digital coded system. The dedicated links do allow the bus to be partitioned into fiber optic and electrical portions. An alternate architecture to 1 would be an architecture which uses dedicated crossed strapped, serial data fiber optic cable to the EMAS, and dedicated serial data fiber optic cable for the sensors which would make as much use of passive fiber optic sensor technology, (reference 19). It would offer little advantage over a fiber optic 2B architecture.

The MUX buses have the next lowest compatibility with fiber optics. Due to fan out power loss and attenuation loss at the couplers, a half-duplex fiber optics bus compatible with MIL-STD-1553 (supports up to 32 remote terminals) cannot be built with present passive fiber optic technology. Future developments in transmitters, receivers and couplers might make this type of bus available; however, the risk factor is high and the cost could be prohibitive.

The architectures with the highest compatibility with fiber optic are 2B and 3B. Both of these architectures use data buses which are composed of multiple, point-to-point, dedicated, serial, digital data links. These links have a single transmitter, a single receiver, and a single cable for transmission and reception in one direction. Fiber optics technology used to implement this type of serial data link is almost mature. Present proposed standards and specifications are under development by the SAE AE-9C Fiber Optics Subcommittee and there will be applications on the AV-8 aircraft.

The architecture with the highest fiber optic compatability are 2B and 3B followed by 2E, 4A, and 1.

9.4.2.8 Resource requirements: Table 28 lists the resource requirements for processor speeds, I/O bus speeds, and interprocessor bus speeds. They were estimated from tables 22 and 23 based on the individual candidate architecture computer architecture and bus topology. The processor speeds are well within 1 million operations per second and the total memory requirement is less than 64K for the flight controls and propulsion functions. These levels are

TABLE 28. - IDEA CANDIDATE ARCHITECTURE RESOURCE REQUIREMENTS

Candidate Architecture / Resource	1	2B	2D	2E	3B	4A
COMPUTERS Speed (KIPS)	FMCS/FCCS/TCCS/FACS 500/70/25/350	FMCS/FTC 500/449	FMCS/FTC 500/449	FMCS/FTC 500/449	FMCS/FTC 500/449	FMCS/FTC 500/449
MAIN MEMORY (16 Bit Words)	215K/15K/10K/30K	215K/55K	215K/55K	215K/55K	215K/55k	215K/55k
I/O BUSES	ARINC 429 Dits	ARINC 429 Dits	MULTIPLEXED Data Bus (1553B)	MULTIPLEXED Data Bus 153B)	MESH Network (Single Link)	MULTIPLEXED Data Bus (1553B)
Speed (Bits/Sec)	10 - 100K	10 - 100K	200K	200K	800K	200K
INTERPROCESSOR BUSES	SERIAL Full-Duplex	SERIAL Full-Duplex	SERIAL Full-Duplex	SERIAL Full-Duplex	SERIAL Full-Duplex	SERIAL Full-Duplex
Speed (Bits/Sec)	500K	500K	500K	500K	500K	500K

achievable with VLSI/VHSIC technology. Since resource requirements are not an issue it will not be included in the evaluation.

9.5 Choice of Preferred System for IDEA Configuration

9.5.1 Preferred candidate architecture selection. - The results of the trade-off analysis allows the candidates to be scored relative to their performance for each of the evaluation criteria. Each of the criteria is assigned a weighting factor which is relative to the criteria's importance in the total cost function. The total grade for each candidate is equal to the following expression.

TG(i) = Summation (i = 1,8) of [WF(j)*GRD(i, j)]

TG - total grade
WF - weighting factor
GRD - raw grade
i - index for each of the candidates

1 - Candidate Architecture 1
2 - Candidate Architecture 2B
3 - Candidate Architecture 2E
4 - Candidate Architecture 3B
5 - Candidate Architecture 4A

j - index for the eight criteria

1 - Reliability WF = 10
2 - Certificability WF = 8
3 - Invulnerability WF = 10
4 - Acquisition Cost WF = 6
5 - Weight Related Cost WF = 7
6 - Technical risk WF = 8
7 - Extended Development Capability WF = 3
8 - Fiber Optics Capability WF = 3

Each weighting factor is ranked from 0 to 10 where 0 corresponds to the least important and 10 corresponds to the most important criteria. The following scale is used to assign raw grades to each of the candidates for each of the criteria.

1.	Reliability	0 - low	.	10 - high
2.	Certificability	0 - low	.	10 - high
3.	Invulnerability	0 - low	.	10 - high
4.	Acquisition Cost	0 - high	.	10 - low
5.	Weight Related Cost	0 - high	.	10 - low
6.	Technical Risk	0 - high	.	10 - low
7.	Extended development Capability	0 - low	.	10 - high
8.	Fiber Optics Compatibility	0 - low	.	10 - high

For this evaluation the highest raw grade and total grade is the most desirable grade and indicates the optimum candidate out of those evaluated.

Table 29 lists the raw grades and weighted grades for each candidate architecture and evaluation criteria. The total grade normalized by the sum of the weighting factors is shown at the bottom of the matrix. The architecture with the highest grade, the preferred architecture, is Candidate Architecture 3B, the Centralized/MESH II architecture.

9.5.2 Preferred IDEA digital flight control system description. - The Centralized/MESH II architecture has been selected as the preferred IDEA flight control system architecture based on the evaluation methodology used in Section 9.5. This architecture has been described in Section 9.2.5 and Figures 111, 118 and 119. Although sufficient explanation and justification was given for this architecture in that subchapter, the outstanding features of the MESH will be summarized:

- centralized DFCS architecture
- the centralized computer is located in a pressurized, cooled compartment
- propulsion and flight controls functions are integrated into the same computer
- centralized, quadruplex fault tolerant computer
- frame synchronization is used for the quadruplex processors
- serial interprocessor buses
- dissimilar digital backup processor protects against generic software errors
- I/O data bus is a reconfigurable damage/fault-tolerant MESH network
- switched network technology developed by Charles Stark Draper Labs
- network accomplishes path switching and device and data multiplexing using a command response protocol
- centralized bus control is resident within the fault tolerant processor
- remotely located nodes can be closely attached or embedded high temperature electronics
- the network is composed of two independent network strata of 16 nodes each
- the switchable nodes are connected by full-duplex, bidirectional serial data links

TABLE 29. - IDEA FLIGHT CONTROLS TRADEOFF MATRIX

Evaluation Criteria	Weighting Factor	1		2B		2E		3B		4A	
		Grade	Wtd Grd	Grade	Wtd Grd	Grade	Wtd Grd	Grade	Wtd Grd	Grade	Wtd Grd
Reliability	10	8	80	0	0	8	80	10	100	9	90
Certificability	8	10	80	10	80	8	64	5	40	5	40
Invulnerability	10	5	50	5	50	3	30	10	100	4	40
Acquisition Cost	6	5	30	3	18	5	30	10	60	5	30
Weight	6	3	18	5	30	10	60	8	48	10	60
Technical Risk	8	10	80	10	80	10	80	5	40	4	32
Extended Development Capability	3	5	15	7	21	10	30	8	24	10	30
Fiber Optics Compatibility	3	0	0	10	30	5	15	10	30	5	15
TOTAL	54	353/54=6.54 3rd		309/54=5.72 5th		389/54-7.20 2nd		442/54-8.18 1st		337/54=6.24 4th	

- the point-to-point data links are electrically isolated at the nodes and easily support fiber optics

This is a robust design because the network is divided into two independent strata of 16 nodes each and only one stratum is needed for operational flight. An unforseen hardware failure within the bus hardware could only propagate through one stratum meaning that a single point of failure within the MESH network cannot occur. The two management strategies could be designed differently to avoid generic bus management errors. In addition four direct links from the bus controllers/FTC exist for each group of eight nodes.

9.5.2.1 Electric vs. fiber optic media: The use of fiber optic cable rather than wire can result in a 22 percent savings in cable weight for the FCS. This equates to a reduction of 80 pounds when connector weight is neglected. The block fuel reduction is small compared to the reductions achieved in other systems.

Fiber optics can be implemented if it is shown that the FCS has a high susceptibility to transients resulting from EMI or lightning strikes.

9.5.2.2 Sensors: A skewed sensor arrangement is used such that only 9 sensors instead of 12 are required for double fail-operative capability for each sensor group (rate gyros, accelerometers) for all three axis. Two sensors are located along each axis and three skewed sensors are along vectors on the X-Y,X-Z, Y-Z planes at 45 degrees from the respective axis. Software FDI algorithms detect and isolate sensor failures on all three axis. This results in a reduction of three rate gyros and three accelerometers.

Strap-down ring laser gyro (RLG) inertial reference system (IRS) units are used in place of conventional mechanical gyro units. The cost and weight advantages are small but the reliability is about three times that of the mechanical gyro unit in terms of MTBF hours. The wide bandwidth of the RLG IRS allows the sensor outputs to be used as backup sensors for the flight augmentation.

9.5.2.3 Mesh network technology readiness: Since the Mesh network I/O bus is the one characteristic that separates the preferred architecture from the other architectures, it is necessary to make a general statement regarding the ability of the mesh network technology to meet its technology readiness deadline. The Mesh network technology has not reached technical maturity; however, neither has linear, multiplexed, data bus technology when used for flight-critical applications. The mesh network is not the only one limited by implementation technology such as electrical or fiber optic media, electronic circuitry, or packaging. These limitations are also shared with other architectures such as multiplexed data bus architectures and distributed architectures. Sufficient time is needed to develop the complex bus management software, develop specific designs, and develop assurance methods for system validation and certification. On the other hand, the development of Mesh network technology is accelerating due to ongoing work which involves prototype systems for testing.

Reference 17 documents a network simulation which verifies network management software for a network similar to the preferred architecture network. An actual quadruplex fault tolerant computer was used to run the software and manage the network. The results demonstrated that a 4,000 byte program could successfully grow, manage, and reconfigure a 16 node network through one bus controller.

Reference 20 documents an iron bird simulation of a mesh network interfaced with an A-7 simulator. The flight computers and mesh network bus were implemented in hardware and fault testing was performed with favorable results.

The advanced information processing system (AIPS), under development by Draper labs, proposes to use a five port node Mesh network for all I/O and interprocessor buses. The AIPS system was designed to meet the information processing requirements for a fly-by-wire commercial transport aircraft, tactical military aircraft, and manned and unmanned space platforms. The system is in the preliminary design stage and will be built and tested on AIRLAB at NASA LARC. Phase II of the AIPS program which calls for building and testing a prototype system is scheduled for completion by March, 1986. Reference 21 documents the AIPS system specification which includes the Mesh network node design.

9.5.3 Flight control system input for ASSET.

9.5.3.1 Aquisition cost and weight data: Aquisition cost and weight for the preferred architecture wire, computers, and bus hardware have been computed and listed in tables 26 and 27. This was added to the cost and weight of the remaining IDEA DFCS equipment listed in table 22 and loaded into the ASSET flight control data input.

9.5.3.2 Maintainability: Maintainability costs enter into total life cycle cost data which are evaluated by ASSET. Life cycle cost were not used for the candidate architecture selection but maintainability data was generated for the preferred architecture for input to ASSET.

The maintainability aspects of the key components comprising the selected architecture is shown in table 30. The man-hour values are predicted on the components being located in readily accessable avionics bays where feasible. This eliminates the need for using workstands and removing access panels. Certain components, e.g., engine controllers, require the use of workstands and gaining access through cowl doors. These factors have been considered where appropriate.

9.6 Identificationn of Critical Technologies

The following critical technologies for the IDEA aircraft must be available by 1990.

- Analytical Redundancy - Analytical redundancy is the calculation of desired sensor signals using different sensor inputs and known mathematical relationships. Currently these calculated signals are used for

TABLE 30. - PREFERRED IDEA FCS MAINTAINABILITY

Component	Qty	MH/FC*	MH/FH*	Matl $/FC*	Matl $/FH*
Throttle Quadrant	1	49.1	163.8	912.6	3,042.0
Column LVDTS	2	5.7	5.7	336.0	336.0
Wheel RVDTS	2	5.7	5.7	448.0	448.0
Pedal LVDTS	2	5.7	5.7	336.0	336.0
AOA Sensors	4	844.8	4,224.0	16,256.0	81,280.0
Side Slip Sensors	4	844.8	4,224.0	16,256.0	81,280.0
Discrete SW. Assy.	2	39.4	785.4	396.0	7,920.0
EFI	2	3,281.2	3,281.2	75,000.0	75,000.0
FAS CDU	2	2,898.0	4,830.0	16,320.0	27,200.0
FCS CDU	2	2,898.0	4,830.0	16,320.0	27,200.0
Maint/Test Panel	1	562.5	562.5	6,800.0	6,800.0
Thrust Rating Panel	1	308.0	616.0	3,400.0	6,800.0
Variable Feel Units	2	147.0	2,940.0	1,435.0	28,700.0
Parallel Trim Servo	6	283.5	5,670.0	4,305.0	86,100.0
Pitch AP System	2	1,224.9	2,041.6	22,828.8	38,048.0
Roll AP System	2	1,224.9	2,041.6	22,828.8	38,048.0
YAW AP System	2	1,224.9	2,041.6	22,828.8	38,048.0
Rate Gyros	12	4,073.2	4,525.7	10,022.4	11,136.0
Accelerometers	12	3,056.9	3,396.6	87,112.8	96,792.0
Spoiler EMAS	12	121.2	12,117.6	1,449.4	144,936.0
Aileron EMAS	10	198.9	19,890.0	2,340.0	234,000.0
Wing Tip Accel.	4	1,076.3	1,195.8	29,037.6	32,264.0
Hor. Stab. EMAS	4	83.5	8,353.8	936.0	93,600.0
Rudder EMAS	3	65.0	6,499.4	702.0	70,200.0
Engine 1 & 3 Contlrs.	2	322.0	6,440.0	2,800.0	56,000.0
Engine 2 Contlr.	1	172.5	3,450.0	1,400.0	28,000.0
IRV	3	2,520.0	2,520.0	33,000.0	33,000.0
MESH Nodes	32	1,177.6	1,777.6	8,704.0	8,740.0
TOTALS		28,689.0	107,780.0	404,470.0	1,357,400.0

$*X10^6$ MH/FC = Man Hours/Flight Cycle MH/FH = Man Hours/Flight Hour

Maintenance costs, MC, per flight is:

MC = MH/FC + t x MH/FH + Matl $/FC + t x Matl $/FH

where t = flight time in hours.

comparison monitoring purposes to aid in isolating a final failure of a dual sensor. It has been demonstrated, however, that the reconstruction of certain sensor signals can be of sufficient fidelity to be used for control law computations. In this case some degree of stability margin enhancement may become necessary. Analytical redundancy may be applied to the sensor set of the IDEA aircraft to reduce the sensor complement for certain sensors or may also be used to increase the redundancy of other more critical sensors. Analytical redundancy techniques are just beginning to be applied to production systems in the area of engine controls. There is still some disagreement as to the bandwidth of the estimated sensor signals and the problems this may cause with the failure monitoring thresholds and active control functions.

- VHSIC/VLSI - Microcircuits for microprocessor chips (MIL-STD-1750A) and serial data bus chips (MIL-STD-1553B) will be available in two or three chip sets by the late 1980's. One prediction is that ultrareliable embedded processing components will be available in the late 1980's to support digital calculations for flight controls. This increase in reliability will be derived from increased integration with reduction of chip counts and connections between chips, and also by fault tolerance and self-testing mechanisms embedded on the chips. Chip costs will be reduced slightly. However, systems containing these new devices will not necessarily be low in cost due to increases in system redundancy as a result of fly-by-wire design. There is also a trend toward increased functional integration which tends to increase costs.

- High Temperature Electronics - Research and development in high temperature electronics and in high temperature packaging techniques has psignificantly since work was started in the 1960's by NASA for application to spacecraft. Research into semiconductor components involving high temperatures is on-going at several companies and government organizations. These activities can be grouped into the following categories:

  1. Developments for relatively short-term reliability as for geothermal and oil drilling applications.

  2. High-temperature testing to accelerate failures which would occur at normal temperatures, as is done by Bell Laboratories and others.

  3. Those which are being conducted under government sponsorship, such as the NAVAIR/NRL/GE program for application to engine controls and work being performed at NASA Lewis.

- Research in packaging technology has been performed by HR Textron using existing components. Hybrid techniques have been employed wherein semiconductor chips were attached to the package substrate with eutectics, reducing the thermal path impedance between the microcircuit package and the semiconductor junction from approximately 90°C/watt to 10°C/watt. This work has been done by the General Electric Company at their Evendale Engine Division in applications related to the FADEC

engine control program. High temperature electronics will be applicable to the IDEA flight control system in the areas of bus interface electronics, engine control, and smart servoactuators. These applications call for electronics to operate in uncontrolled environments within the aircraft which experience temperature extremes during normal operation.

- Fiberoptics - Fiberoptic data communications offer three basic advantages over conventional cables:

  1. Greater bandwidth over longer distances.

  2. Improved physical characteristics (e.g., lighter weight).

  3. Increased channel integrity (e.g., EMI immunity).

  The unique physical characteristics of fiberoptic cables offer significant benefits. They are small in size and very light in weight. In addition since fiberoptic cables contain no conductive materials, they are not potential current paths, do not require any grounding, and cannot cause short circuits. The last feature is of importance if the cables must be routed through potentially explosive areas (e.g., fuel tanks). Finally, fiberoptic cables are not affected by electromagnetic interference (EMI) produced by power distribution, relay switching, or lightning. From the above it can be seen that fiber optic communication has considerable appeal for application to the IDEA aircraft. The EMI characteristics, light weight, and high bandwidth will be of considerable value to the advanced flight control system architecture of the IDEA aircraft.

- Programming Languages (Ada) - Ada provides a significant advance in program language design and capabilities. Ada is the result of the Department of Defense's (DoD) effort to create a single, common higher order language for implementing military embedded computer systems. The features of Ada were specified to meet the requirements for programming language with considerable expressive power covering a wide variety of applications from large ground-based systems to airborne flight-critical systems. The applicability of Ada to the IDEA aircraft flight control system will derive from the effort expended by the DoD to provide program development support environments, efficient run-time libraries, and the reuseable software that is generated for flight controls application. By applying reuseable software the certification tasks associated with the software may be reduced.

- High Speed Data Buses - Depending on the amount of information to be transferred or the rate at which it must be updated, there will be a need for a high speed data bus operating at speed of 10 MHz or higher. The SAE AE-9B subcommittee on high speed data buses (HSDB) has begun efforts to replace the current MIL-STD-1553B (1 MHz) data bus with a higher speed bus that would be adequate for future integrated system requirements. This new higher speed information transfer system (ITS)

will employ an operational protocol that will provide the various systems on the bus and their sensors with independence and fault isolation, as well as distributed control of the common data bus. This last characteristic, distributed control, is a major departure from the current MIL-STD-1553B data bus, which uses a command response protocol. Due to the increased integration within the flight control system a high speed data bus will have broad application to the IDEA aircraft. A bus of the type currently under consideration would be applicable to both the MESH network and the more conventional parallel redundant bus implementation. In addition a high speed bus would be applicable to the power control system as well.

- Software Fault Tolerance - Fault-tolerant software design techniques have been developed to be used in the design and development of software and hardware, which itself will be tolerant of software faults. A software fault is defined as a design defect in the introduced during the software development cycle. The appearance of a software fault places the system in an erroneous state which may cause the system to fail. Software faults may be due to incorrect specification, incorrect algorithm, incorrect logic, coding mistakes, etc. Approaches to software fault tolerance include n-version programming, recovery blocks, and hybrid techniques. Software fault tolerance will have a significant impact on the IDEA flight control system due to its heavy reliance on software-based digital systems. As software fault tolerance techniques mature they will help bound the verification and validation tasks necessary for certification.

- Fault-Tolerant Processors - Considerable work has been done improving the reliability of digital flight control computers. Three advanced fault-tolerant computer systems are currently under development for possible application to digital flight control systems. These are:

  1. Software implementation fault tolerance (SIFT)

  2. Fault-tolerant multiprocessor (FTMP)

  3. Multimicroprocessor flight control system (MFCS)

  The first two fault-tolerant computer concepts achieve fault tolerance through reassignment of tasks among available processors. Task reallocation is necessary because the two systems are based on the use of multiple processors operating in parallel to obtain the required throughput. If a processor performing a critical task fails, another must be reconfigured to take its place. The primary differences between SIFT and FTMP is the method of detecting failures. SIFT relies on software implemented fault detection, whereas FTMP utilizes hardware bit-synchronization. Each of these approaches has unique problems associated with it. Since SIFT utilizes software for fault detection the software must be error free which complicates the validation tasks. FTMP, due to the reliance placed on bit-synchronization, must ensure that all processors are in lock-step, which requires a fault-tolerant

master clock. Solutions to these problems have been proposed by the associated contractor.

The Multimicroprocessor Flight Control System (M^2FCS) is an ultra-reliable flight control system developed by the Honeywell Systems & Research Center. The design of the M^2FCS is based on two conceptual building blocks, the self-checking pair and the information transfer system. The self-checking pair concept uses processors and bus pairs as the basic unit of redundancy. Each processor and bus in the system is paired. The halves of a pair perform identical functions on identical inputs. Non-identical outputs disconnect the pair from the rest of the system. The information transfer system is tolerant of its own internal faults and is protected against external hazards. The information transfer system also allows the consistent exchange of data.

- Bus Network - Mesh bus networks are major variants of standard multiplex systems. The topology of mesh networks is one in which subscribers contain, or adjoin, repeater and switching circuitry referred to as nodes. Each port interfaces one end of a link. All links are full-duplex so that commands can be sent to reconfigure the network despite the presence of anomalous transmissions from a subscriber to a node.

  From this basic ability to reconfigure the bus routing high-survival chacteristics of the network are derived. Note that once a bus has been created, it operates exactly as a true bus using standard bus protocols. Thus, there are no operational overheads associated with the operation of the virtual bus beyond those imposed by a standard bus and an initial setup or confiuration procedure.

  A mesh data distribution network will provide for highly reliable data transfer for the IDEA flight control system. In addition to providing high reliability, the mesh network, if provided with adequate spare nodes and links, may provide the capability of dispatching the aircraft with a known failure within the data distribution system. This would lower maintenance costs by allowing an increase in the time between maintenance and also reduce operational costs by being able to dispatch with an existing failure.

- Surface Segmentation - Surface segmentation is a form of redundancy where a particular surface (i.e., elevator) is segmented (split) to increase the probability of operation after a failure. Generally surface segmentation is used to increase the survivability of tactical aircraft against weapon hits. If one of the segments of a surface receives a weapon hit or if the actuator should jam or fail then the other segment will still provide a limited amount of control power. The aircraft will still be controllable but with a certain amount of degradation. Another advantage of surface segmentation is that it simplifies the actuator interface to the surface since the system has more tolerance to an actuator jam. Surface segmentation may be applicable to the IDEA aircraft for preventing actuator jam from disabling a

particular control axis. This is an area that will be driven by cost trades between actuator complexity for a single surface and the reduced complexity for a segmented surface.

- Dissimilar Redundancy - Dissimilar redundancy is applied to avoid a generic fault from disabling a system by simultaneously failing all channels at once. Dissimilar redundancy may be applied to both hardware and software systems. A current example of dissimilar redundancy is the Space Shuttle which uses an independent fifth backup computer programmed in a different language from the four primary computers. Dissimilar redundant systems have been developed and may be used for implementing essential subsystems for IDEA.

- Assurance Technology - Assurance technology consists of all facets of showing (assuring) that the flight control system is safe and meets the requirements of FAA Advisory Circulars 1309-1 and 20-115. Currently aircraft software based systems are developed and validated under the guidelines of RTCA DO-178 which outlines a development process for the operational software. This document currently only addresses flight-phase critical systems (i.e., Autoland) and will have to be expanded to cover full-phase critical systems such as relaxed static stability augmentation systems. Work in the areas of design methodologies, verification testing, formal proofs of correctness, and bounding the validation tasks must be addressed for the introduction of the IDEA aircraft.

- Integrated Maintenance Concepts - Integrated maintenance concepts will utilize the increased computational power of digital flight control systems to enhance in-flight monitoring of the total flight control system. Positive identification of failures or transients in flight will reduce maintenance time in taking corrective action to repair the flight control system. These systems will be able to exploit advanced technology to isolate failures to the card level and possibly to the chip level within the IDEA flight control system. Included within these systems or within the ground support equipment will be expert systems to aid maintenance personnel in diagnosing problems and taking corrective action.

9.7 FCS Plans and Resources for Technology Readiness

To ensure the timely availability of the benefits realizable through IDEA configurations, it is essential that certain FCS technology needs to be addressed responsively in the near term. This entails the definition, justification, and execution of an FCS technology readiness plan that ultimately permits the safe and dependable implementation of full-time flight-critical DFCSs. In turn, this necessitates careful prioritization of need, as well as the recognition of existing technology initiatives, to direct the proper allocation of research resources.

The previous section described the preferred IDEA FCS architecture and disclosed its composition, while section 9.2 identified certain tradeoffs that refine its attributes. Crucial among the latter are implementation risk and ease of validation which expand into an array of technology needs. Since the background for these needs is discussed in the next section, the intent here is to define a strategy for fulfilling them.

9.7.1 FCS enabling technology needs for IDEA. - Basically, sufficient practitioner-oriented technology must be secure to design, implement, install, and validate IDEA FCSs on a consistent, dependable, and credible basis, for fulltime flight-critical DFCSs. This needs to done in a systematic, highly coordinated manner. As a unifying point of departure then, the first need is assurance-driven system development methodology that fosters, enables, and resolves system failure rates of 10^{-9} or less. It should also facilitate system integration on a technically mature level.

The major and as yet unresolved concerns over achieving such a methodology are the quantification of software system reliability, the overall adequacy of fault tolerance, and system robustness in the installation environment. These concerns in turn expand out into the more specific issues given in figure 123. For the present, however, the higher-level concerns merit further assessment in the context of a total system development methodology.

System reliability - Demonstrating compliance with FAA Advisory Circular (AC) 25.1309-1 for critical system function would be currently a virtually intractable undertaking for a DFCS because of inability to characterize the impact of software discrepancies. Existing software reliability models have not proven very accurate or consistent, nor have they been able to capture the effects of combined hardware/software interactions. Since even the promise of acceptability for critical applications does not appear to exist, entirely new approaches and methods would seem proper. In any case, software system reliability constitutes an urgent IDEA need.

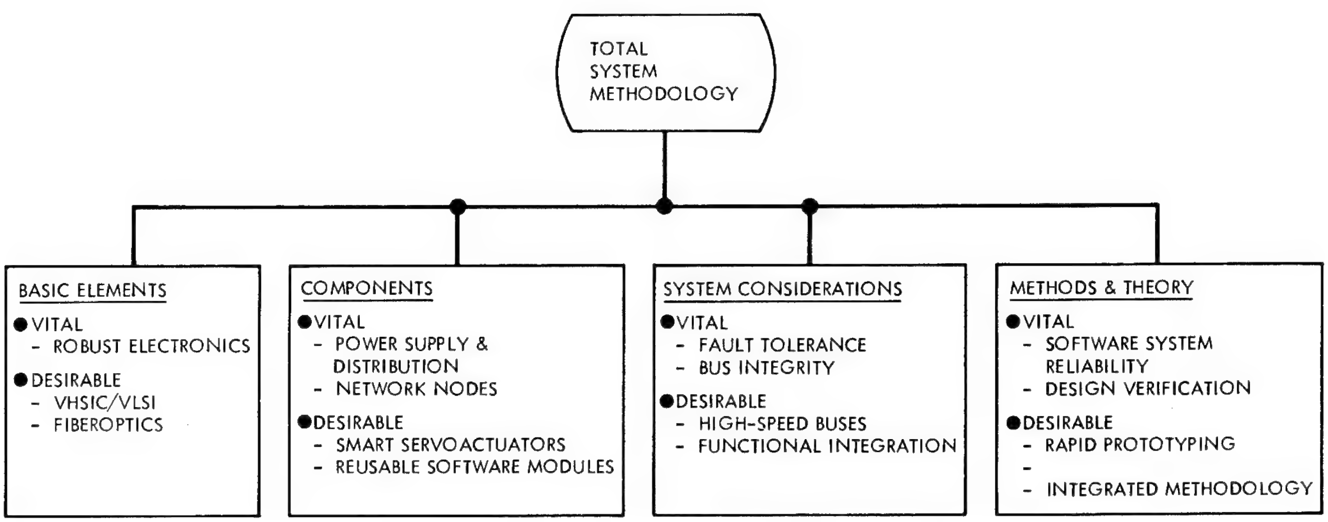


Figure 123. - Enabling techology needs.

Fault tolerance - Not only must fault tolerance be devised and validated primarily at a system level, but it must be carefully tailored to particular applications. System resources must be allocated in a manner that affords an overall balanced fault protection. The dominant concern in redundant DFCSs at present, however, is strictly software based; this is the possibility of a generic software error that coincidentally appears in all channels. Since all channels would manifest the same anomalous behavior the system could not detect or compensate for the error.

Some form of dissimilar software is required to achieve tolerance of software faults. This could take the form of dissimilar software in different computational channels or a type of software fault tolerance such as n-version programming. In either case, the software to achieve this has yet to be developed. This results largely from the need to maintain close coordination among wide-bandwidth, redundant channels. The enabling technology is particularly weak in the case of fault-tolerant software which is very application dependent.

As evident in the IDEA reliability analyses, major leverage for system improvements lies in better electrical power distribution and control systems, as well as in upgraded electronic power supplies. The former issue is most acute in the case of an all-electric aircraft, and the latter in the case of a safety-critical system. Both concerns need to receive some of the emphasis currently accorded to electrical power generation.

Environmental robustness - Digital flight systems have been rather susceptible to aspects of the environment within an aircraft including temperature extremes, humidity, and vibration, electrical power transients and electromagnetic interference (EMI); and poor heat sinking at device mounting points and exposure to inadvertant damage by humans.

The environmental implications for the electronics are particularly formidable. Basically, there is a need for certain quasi-standard components that can be servicable over a range of installation conditions. For example, the same type of mesh network node should be dispersed over the aircraft, conceivably with temperature extremes being encountered simultaneously in the nacelles and in the wings. The high temperature concerns may be further be accentuated by components being mounted on composite materials which are generally thermal insulators.

Thermal cycling, moreover, is of concern with regard to fiber optic conductors, which tend to exhibit increased attenuation over curved spans due to differing increments of expansion/contraction. Susceptibility to electrical power transients is another concern, and it applies to both electro-optic converters in fiberoptics systems and electrical bus media. This is only one aspect of the concerns regarding the integrity of data buses which must incorporate means to withstand or recover from transient disturbances.

Validation conclusiveness: The conclusiveness of system validation depends on all three of the foregoing technology needs coupled with other assurance technologies. The latter are of course the subject of concern here. For clarification, system validation here is used in essentially the same

meaning as FAA certification, and hence validation activities are considered to begin long before the onset of actual validation testing. Technology needs as derived from IDEA FCS requirements, moreover, indicate that validation activities must begin at the initiation of the system development cycle, i.e., when FCS requirements are being defined.

The pivotal assurance technologies that focus on system structure, as opposed to system function, include rapid protoyping, design verification, and software complexity metrics.

Briefly, rapid prototyping is needed to confirm user requirements and to alleviate technical risk at the outset of the development cycle. This becomes increasingly important as DFCSs become more sophisticated and safety-critical. Rigorous design verification is essential to ensure the existence of and compliance with an explicit and precise design solution. Only then can system implementation begin, and only then is there an adequate basis for subsequent system validation. Aside from design utilitiy, software complexity metrics offer a possible basis for quantifying validation test requirements and coverage actually achieved.

9.7.2 FCS technology readiness plan. - Many diverse technology initiatives promise to contribute to the realization of IDEA-oriented FCSs. In some cases, the motivation is somewhat different from the present application, but the intended results may translate well into the IDEA context. To avoid redundant R&T efforts it is essential to acknowledge such programs. It should be noted, however, that while all of the IDEA-essential enabling technologies are currently being investigated in some sense, there still remains the need to formulate certain new projects to ensure fulfillment of the DFCS technology needs for the realization of IDEA configurations in the early 1990s.

A summary of prioritized technology needs is presented in figure 123, along with suggestions regarding R&T emphasis. To indicate interrelationships and interdependencies among technology elements, figure 124 depicts a technology roadmap to provide the desired technology readiness for IDEA-oriented DFCS. Refinement and implementation of this plan of course involves considerable definition beyond that offered here, but that presented is deemed to be a sound nucleus for follow-on R&T activities.

9.7.3 Technologies appropriate for government support. - Figure 125 shows a list of technologies which are appropriate for government support along with schedules and resources. The technologies are indeed high risk due to their lack of maturity compared to other technologies. Unlike other technologies, (energy efficient engine, advanced electrical power systems, electromechanical actuation, etc.,) the development of these DFCS enabling technologies has not been driven by economic incentives which have promoted development of other technologies by private industry. However, these critical path technologies will enable high payoffs. Without them the IDEA aircraft would be unachievable.

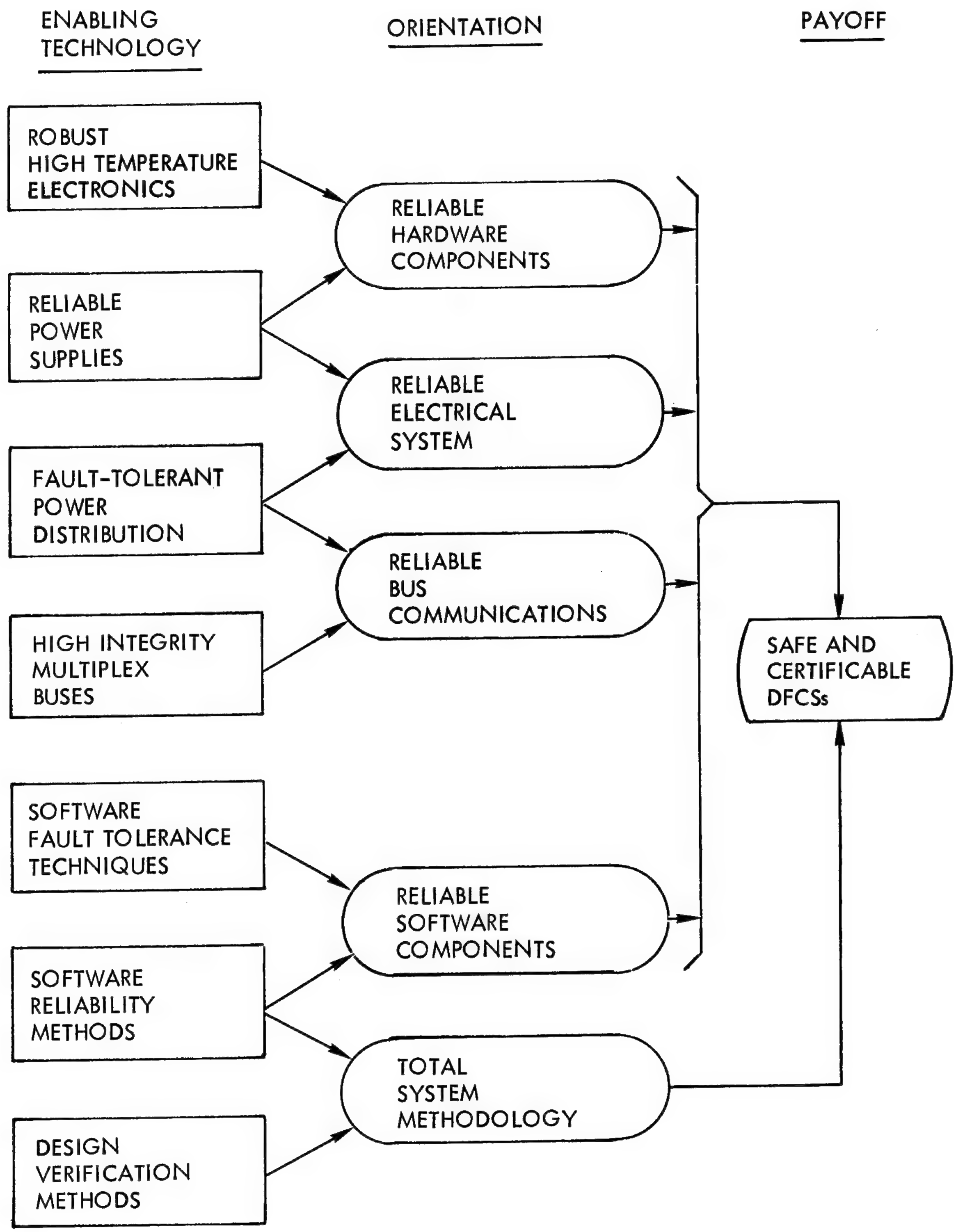


Figure 124. - DFCS enabling technology roadmap.

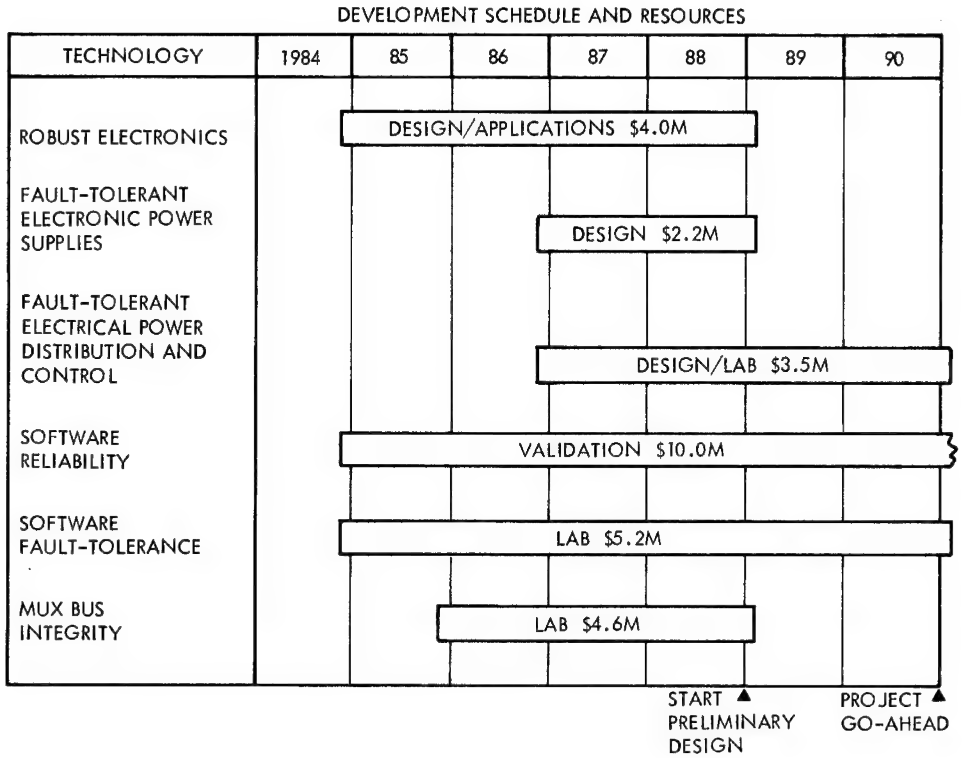


Figure 125. - Technologies appropriate for government support.

Robust electronics and fault-tolerant power supply technology development should be mature prior to 1988 so that preliminary design can proceed on schedule. Fault-tolerant electronic power supplies incorporate fault-tolerant concepts with robust electronics and its development will lag robust electronics by two years.

Fault-tolerant electrical power distribution and control will also lag the development of robust electronics by two years. This program would consist of designing power distribution and control devices suitable for flight-critical electrical power systems, design of fault-tolerant distribution and control algorithms, and laboratory testing to validate a prototype system and verify all concepts and devices.

Software reliability will mainly involve validation of DFCS and electric power system software to assure reliable software. Its development will proceed from 1985 through to certification since it is not design oriented. Software fault-tolerance will proceed from 1985 up to project go-ahead in 1990 at which time a decision must be made regarding the desired software fault-tolerant technique and the processor requirements for that technique.

MUX bus integrity mostly involves laboratory testing of a representative IDEA DFCS bus to determine its integrity in a high EMI environment or in the presence of a lightning strike. It will also be necessary to evaluate the decreased protection due to use of composite structures. The evaluation should be completed by the beginning of the preliminary design phase so that

decisions regarding system protection, communications media, etc, can be made. This development will lag robust electronics because the desired level of MUX bus integrity will be dependent to some extent on the components.

10. AEROELASTIC DESIGN CONSIDERATIONS FOR ALTERNATE IDEA AIRPLANE

In the Alternate IDEA, the aircraft design constraints are relaxed without violating the intent of FAR Part 25 or Part 36 to determine whether synergistic relationships exist between aircraft design and IDEA concepts. The aircraft must satisfy the payload, range and other performance requirements of Section 1.4.2. In this chapter some fundamental changes in aircraft design criteria are discussed. These changes will be possible with the implementation of the IDEA concepts. Some additional advances in the electric secondary power systems (consistent with a post-1994 certification date) are discussed in Chapter 11. The resultant benefits are described in Chapter 13.

The wing design of an advanced transport will incorporate technologies that will be primarily directed to increasing fuel economy. The goal is to optimize the design for minimum drag within the constraints of safety criteria.

The individual technologies to control maneuver loads, flutter, ride quality, reduced static stability, elastic response, gust loads alleviation, and handling quality have been demonstrated by active and passive means. There are no demonstrations of all of these functions integrated in one design process. A paper by Jurey and Radovcich (ref. 22) demonstrated the integration of the active control functions for flutter, handling qualities, gust load control, and ride quality on a supersonic cruise vehicle configuration using just two control system surfaces.

There have been extensive flight experiences with active control systems in commercial transports. The L-1011-500 ACS airplane is currently flying an active control system for maneuver load control, elastic load suppression and gust alleviation. Flutter margin enhancement was demonstrated on an L-1011 test airplane using either the outboard ailerons or horizontal stabilizer for increasing stability of an in-flight low damped mode. The feasibility of reduced static stability was also demonstrated by flight testing.

The rapid development of digital implementation of autopilot functions provides the basis for the conceptual integration of these aeroelastic design functions into a single advanced flight management system.

The design of structures for aeroelastic effects, within the preliminary design schedule constaints, has been making steady progress with the work of Dr. Sobieski of NASA-LaRC in the multilevel optimization approach. One example of the many design tools under development in the industry is Preliminary Aeroelastic Design of Structures (PADS) (ref. 23 and 24).

Active control and flight control systems coupled with the development of advanced flight control computer technology, rapid aeroelastic design of structures, and advanced flight simulation techniques provide bases for the Alternate IDEA airplane proposal.

10.1 Problem Definition

How can aeroelastic design exploit advanced flight management system technology?

Traditional feedback control systems are limited in their ability to reduce structural weight. For example, figure 126 shows that 80 percent of the wing cover weight reduction occurs within 50 percent of the optimum aileron gain for maneuver load control (ref. 24, figure 42).

The Alternate IDEA airplane concept as proposed here must be able to exploit not only the application of the new technology in flight critical control systems but the synergisms of combining active control technology, advanced digital flight control concepts, the all electric airplane technology, and the integration of these technologies into the aeroelastic and performance design. Figure 127 shows the technologies which are required to achieve the goals within the specified design requirement constraints. This figure emphasizes the need to consider the design process as an integrated design activity rather than the design of separate loosely coupled systems. The tools used in preliminary design are listed as ASSET, CADAM, and PADS.

The one factor that has great potential in reducing structural weight for all configurations, especially a large impact on the more refined designs of the future, is the possibility of changing current structural design criteria.

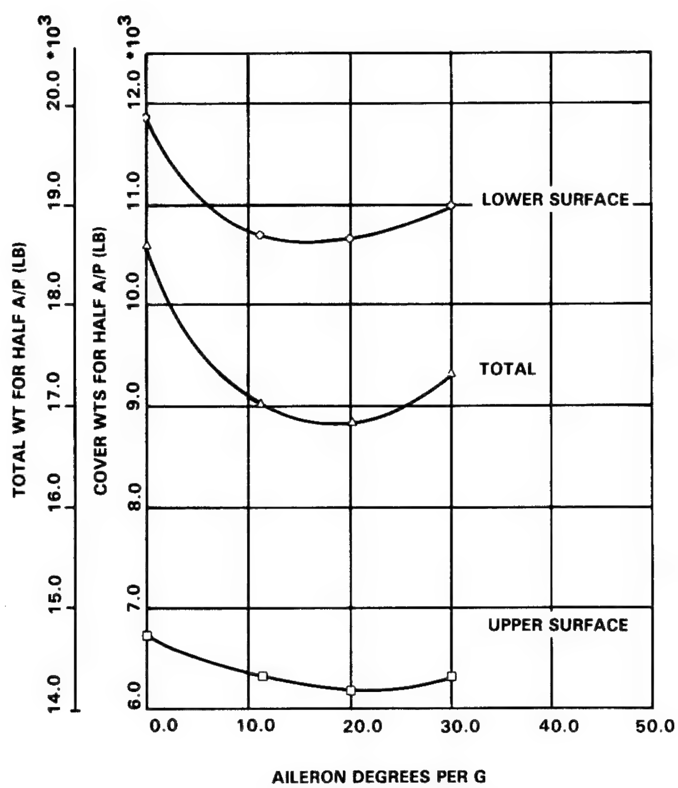


Figure 126. - Wing cover weights for different MLC aileron gains.

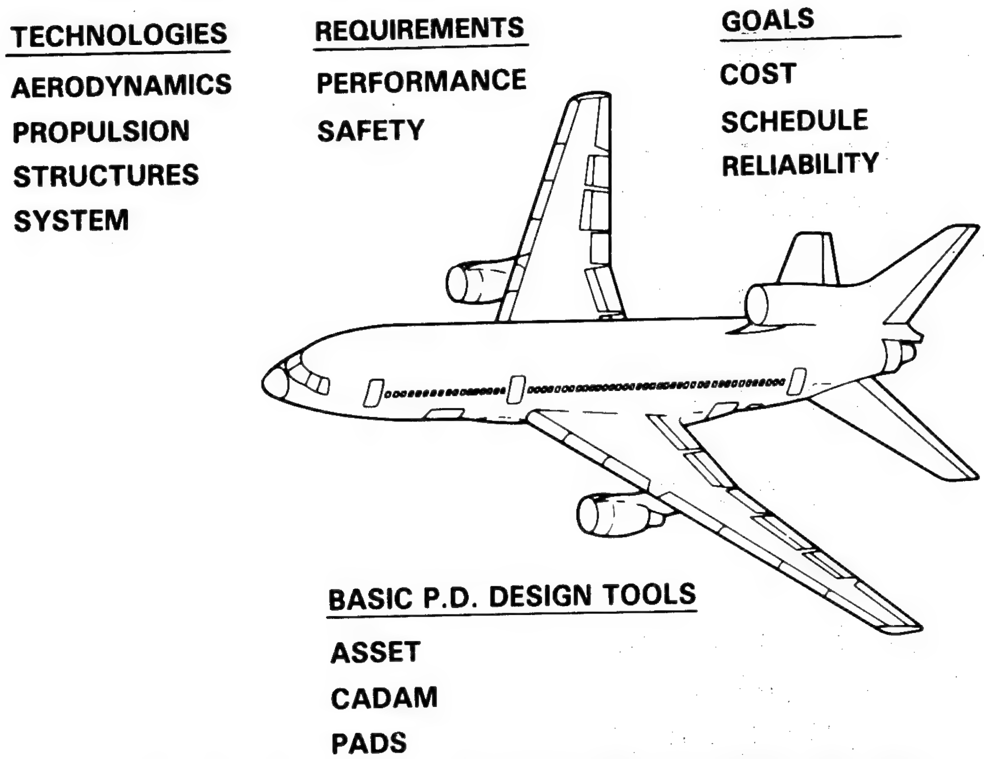


Figure 127. - Integrated preliminary design process.

The role of the advanced flight critical control concepts together with the traditional feedback control systems and advanced sensor technology is to provide the tool to maintain the safety of existing airplanes when existing criteria such as design maneuver load factor are altered.

10.2 Alternate IDEA Airplane Concepts

The approach to an Alternate IDEA airplane wing design will be an integration of active controls, advanced flight control concepts, wing material placement optimization, wing configuration definition, and airplane performance. The Alternate IDEA airplane will have a high degree of computer technology mechanized into the primary flight control systems. The same computers will also serve both the aeroelastic requirements as well as the propulsion functions in one integrated architecture.

The incorporation of the computer technology into the primary flight control systems to reduce significantly structural weight will require high functional reliability associated with advanced arrchitectures.

10.2.1 The central theme. - With the proposed level of computer technology to be integrated in the flight control system the pilot interaction with the airplane control may be open to review. It is possible to think of the computer providing the pilot an airplane platform which is stabilized relative to gust and wind shear disturbances. If the scenarios of gust and wind shear disturbances that generate high g maneuvers and/or hazardous upsets for current airplane designs could be identified, they could be used to define a computerized control system that would lower the maneuver load factor response. This lower level of maneuver g's may form the basis for lowering design loads. This proposal, as summarized in figure 128, is a preliminary look at changing the design criteria to save structural weight.

- **REDUCE THE MAXIMUM LIMIT MANEUVER LOAD FACTOR**

  CURRENT FAR 25 REQUIREMENTS FOR SYMMETRICAL MANEUVERS (A SUBSET)

  −1 TO +2.5 FOR SPEEDS UP TO V_C/M_C

  0 TO 2.5 AT V_D/M_D

  0 TO +2.0 WITH FLAPS EXTENDED

- **REDUCE THE PROBABILITY OF HIGH MANEUVER LOAD FACTORS WITHOUT REDUCING THE LEVEL OF SAFETY PROVIDED BY CURRENT DESIGN CRITERIA**

Figure 128. - Proposal.

However, there are other considerations that enter into the design of the wing besides maneuver and gust loads; namely landing and taxiing loads, ground handling, deflection constraints, etc. An airplane design would be required to focus the integration of various technologies required to evaluate the potential payoffs in these various areas.

10.2.2 Advanced flight control concepts. - The advanced flight control concepts will have additional features over and above the functions attributable to flight critical systems. If the computer integration into the basic airplane operation were to be promoted to a level where the pilot made commands to a computer which in turn configures the control surfaces deflections, engine power settings, etc. to satisfy the flight path requirements, then there would be a means to develop additional concepts that could be critical to the realization of substantial structural weight savings.

10.2.2.1 Advanced flight control manager: The Advanced Flight Control Manager (AFCM) has three concepts to be offered for consideration, namely the expert system, the loads and flight envelope limiter, and the airplane systems resource integrator.

The expert system is resident onboard the airplane and would be available to the AFCM in its stand-alone mode and to the flight crew through some high level language for queries. The expert system would contain predefined actions based on the state of the airplane and its subsystems. It will function much like the flight manual which defines pilot's response to certain observable cues. The expert system could operate online during the flight and inform the flight crew of the systems status and the recommended course of

action. The advantage of an expert system over the flight manual is the access to orders of magnitude more data on the state of the airplane and the shorter time response to define the recommended course of action.

The loads and flight envelope limiting function to be explored within the AFCM architecture is the feasibility of maneuver and gust load limiting to levels related to airplane strength, and envelope limiting related to positive flutter margin requirements. One of the key elements of this function is AFCM knowing what flight condition the airplane is in at all times. For example, weight, center of gravity, airspeed, and altitude at any given time during the flight define the maneuver and gust loads capability of the airplane. The AFCM could take advantage of available airplane strength and flutter margins in responding to an emergency.

The third AFCM concept is the integration of the airplane flight and load control resources to respond to a flight path disturbance when given that authority by the flight crew. Some examples may include:

- the automatic rudder, aileron, engine power settings, and angle-of-attack compensation for engine out condition on take-off.

- the automatic adjustment of airspeed for severe gust penetration requirements, and the adjustment of gust loads control gains.

- the automatic configuration of the airplane to minimize the upset condition due to severe wind shear and the initiation of a recovery procedure.

- the flying of precise trajectories during emergencies such as decompression.

- providing options to the pilot while on emergency is in progress and the AFCM is initiating the first option.

- providing reduced work load in flying the airplane and giving the crew more time to optimize the operation of the airplane.

- integrating collision avoidance information from onboard and ground station inputs.

10.2.2.2 Advanced displays and flight crew interactions: The issue of flight crew interaction with the Advanced Flight Control concepts will not be explored here at any depth. The man-machine interface is not trivial when concepts of an existing operation will probably have to be integrated into the new system. Progress in this area is accelerating, especially in advanced displays. The Space Shuttle Program should provide timely experience concerning the question of the man-machine interface for an aircraft. Also, advanced fighter configurations will address this problem in the near future.

10.2.2.3 Advanced sensor technology: In order to exploit the full potential of active control concepts, it will be necessary to define sensor performance requirements which may not be satisfied with current sensor technology. For example, most current active control load alleviation concepts use the airplane c.g. acceleration as one of the sensed quantities. The load alleviation

benefits are usually local, such as redistribution of wing loads to reduce wing bending moments. With a good lead type sensor the alleviation of airplane response (e.g., airplane acceleration) can be realized by reducing the loads on most or all components on the airplane.

With lead-type sensors, in-house studies indicated that gust loads could be reduced both in the wing and in the horizontal tail. Without lead-type sensors, only the wing loads could be reduced but with significantly larger feedback control gains. Exploration of the Alternate IDEA airplane concepts should include the establishment of advanced sensor requirements, as well as other supporting technology requirements.

10.2.3 AFC and IDEA concepts. - The Advanced Flight Control Concept is illustrated in figure 129. The figure shows that the electric technology may be required to achieve the required reliability and performance. The key to this approach is the AFCM which is separate from the traditional functions assigned to loads control, RSS, and flutter control and takes into account the capability of the airplane as part of the process of doing what is commanded by the pilot. The Advanced Flight Control Manager integrates sensor data, pilot inputs, airplane loads and flutter capabilities, traditional flight management system directives and produces propulsion and control surface deflections that 'best' satisfy built in directives. The Advanced Flight Control Manager will reduce the probability of experiencing high loads due to gust or wind shear by configuring the airplane to minimize potentially large aircraft responses. A possible flight control system configuration is shown in figure 130.

10.2.4 Advanced aircraft configuration designs. - Under Preliminary Aeroelastic Design of Structures (PADS), a number of aeroelasic designs are being generated for high aspect ratio wings with different sweeps. The wing technology design base is the L-1011-500 ACS airplane which was used to check out the current design modules within PADS. This program could serve as bases for evaluating the trade-offs between high aspect ratio wings which produce larger wing weights and decreased drag with t/c and sweep variations.

10.3 Current FAA Policy

Current FAA policy will not allow the full exploitation of the active control technology to control loads and flutter. Along with the design process there must be a parallel study to define the limits of design criteria which would be acceptable to the FAA. The current criteria provide different levels of safety on different structural components depending on the type of loading, e.g., maneuver or gust loads.

The key issue on criteria changes is the concept of equivalent safety. It must be proven that airplanes designed to new criteria will have safety equivalent to existing airplanes. By investigating the different types of loading and defining the criteria such that each type provides a statistically equivalent level of safety, excess structural weight can be removed without affecting the level of safety provided by the current criteria. Maneuver design criteria, based on statistical data and mission analysis, similar to gust design criteria currently in use, is a means of arriving at consistent criteria for different types of loading.

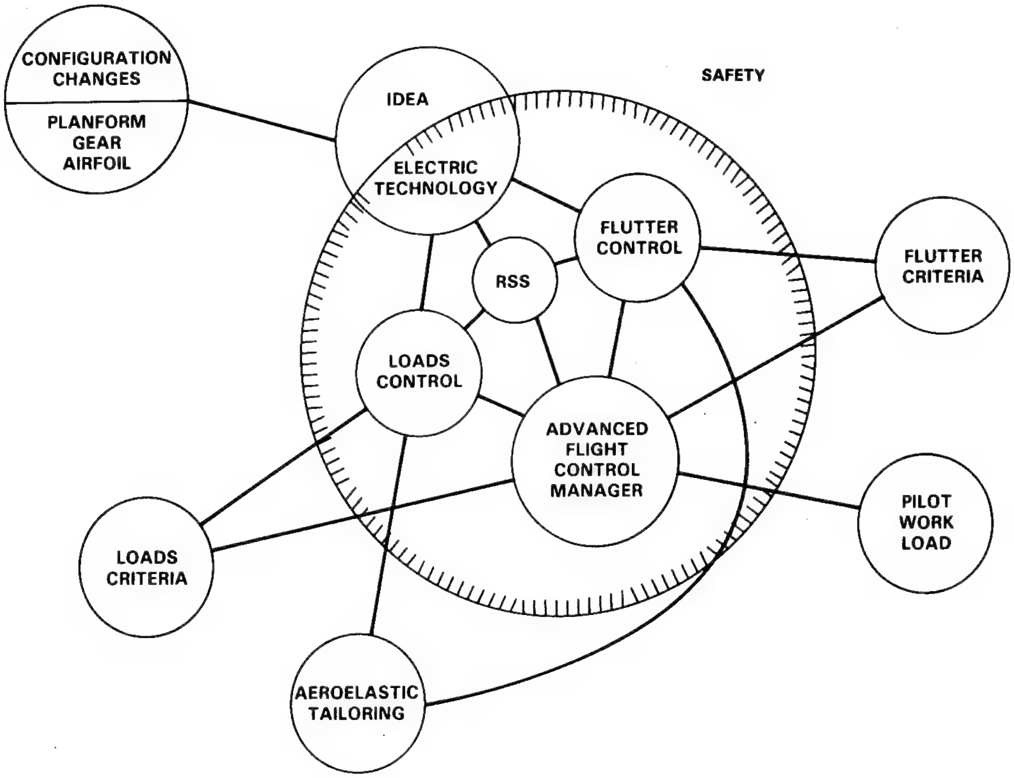


Figure 129. - Advanced flight control concept.

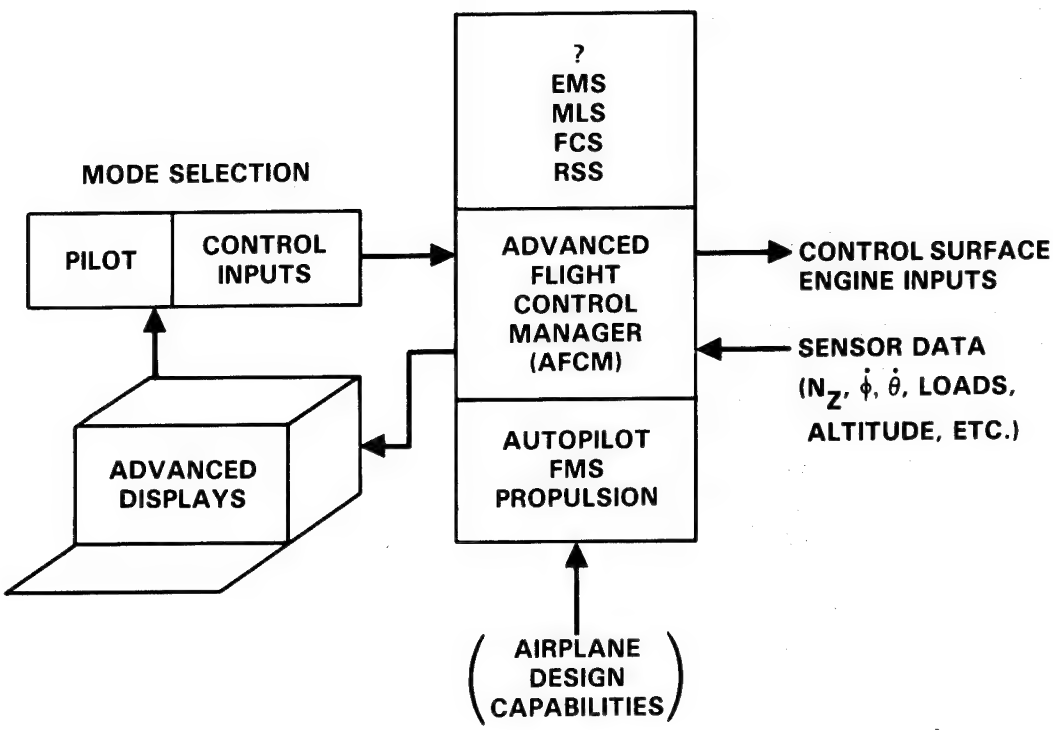


Figure 130. - Flight control system configuration.

10.4 Quantitative Results and the Payoff

At this stage, it is almost impossible to establish what will be the quantitative results if active control, aeroelastic tailoring, advanced computer technology, supercritical airfoils, and new design criteria were integrated into a design of a new wing. Flutter prevention and high gust loads, if a problem, may not be the only major problems of a high aspect ratio wing design. The proposed means to exploit the potential of a design which incorporates this advanced technology is to focus on a particular family of designs where trade-offs are made and critical design elements identified.

One possibility is the relaxing of the 2.5 g requirement for maneuver loads. In order to achieve this relaxation a number of issues must be investigated to determine the feasibility and level of reduction. However, as a first cut at the potential payoffs, an arbitrary reduction of the 2.5 g maneuver limit load requirement to 2.0 g was made and a number of sensitivity wing design calculations were processed through the PADS program (ref. 24).

The reference aircraft (for this exercise, although not for the IDEA configuration) has an aspect ratio of 7.63 and a sweep of 35 degrees. The design technology base incorporates the L-1011-500 ACS requirements. PADS aeroelastic model for the transport configuration includes the sizing of the cover panels for buckling and stress. The sizing program uses the same criteria as was used in the L-1011 certification calculations. The structural model is a full 3-D finite element model and sizings included panel covers for upper and lower surfaces along four chordwise bays, as shown in figures 131 and 132. The high aspect ratio wing design did not include dynamic gust loads. The study included maneuver load control and aspect ratio sensitivity computations.

Figure 133 summarizes the results for an aspect ratio 12 wing with 35° sweep. The wing weight reduction was 11.7 percent when the maneuver limit was reduced from 2.5 g to 2 g. The reduction in weight due to maneuver load control was 5.1 percent.

The reduction in wing weight when the IDEA configuration (25° sweep) is redesigned for a 2 g maneuver limit is very similar. The resulting reduction in block fuel and DOC is described in Chapter 13.

The relationship between PADS and ASSET is shown in figure 134.

10.5 Proposed Study Plan

The objective of the proposed study plan is to mature concepts which have been proposed for the Alternate IDEA airplane into a technology database which could offer design options for the development of an IDEA configured transport aircraft. The proposed study has four segments. The strawman plan defining the overall time schedule and the interaction of the four segments is shown in figure 135.

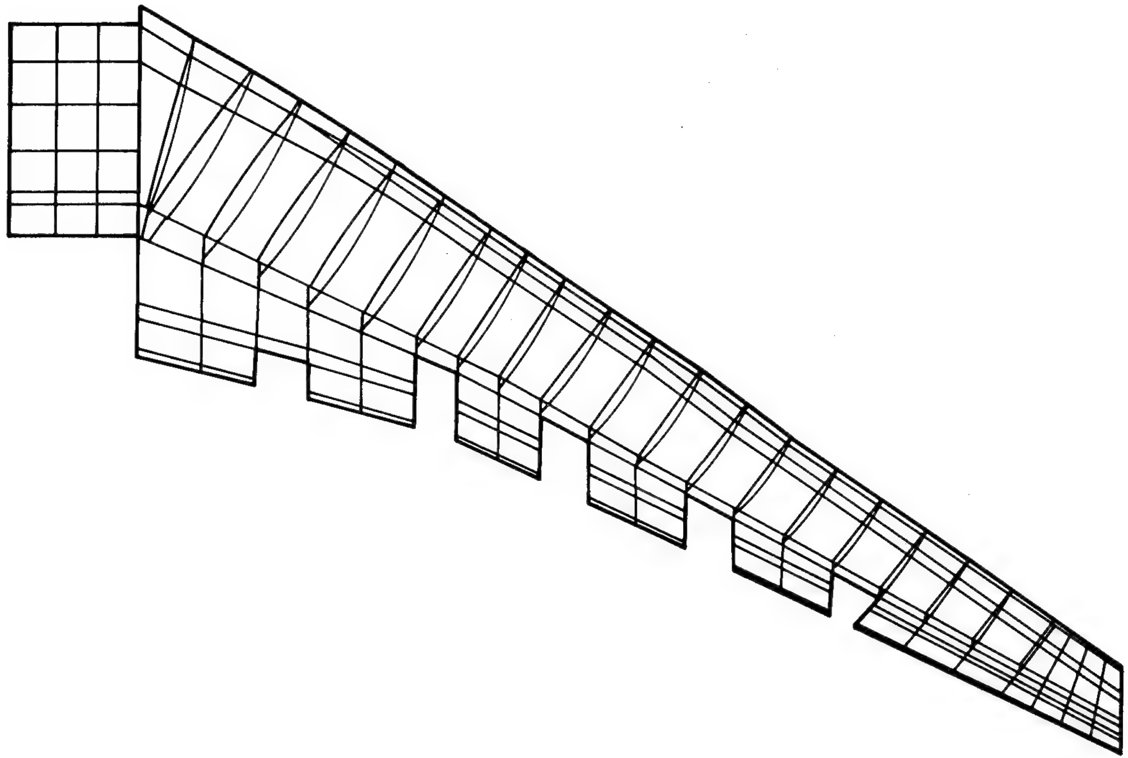


Figure 131. - Wing section 3-D structural finite element model.

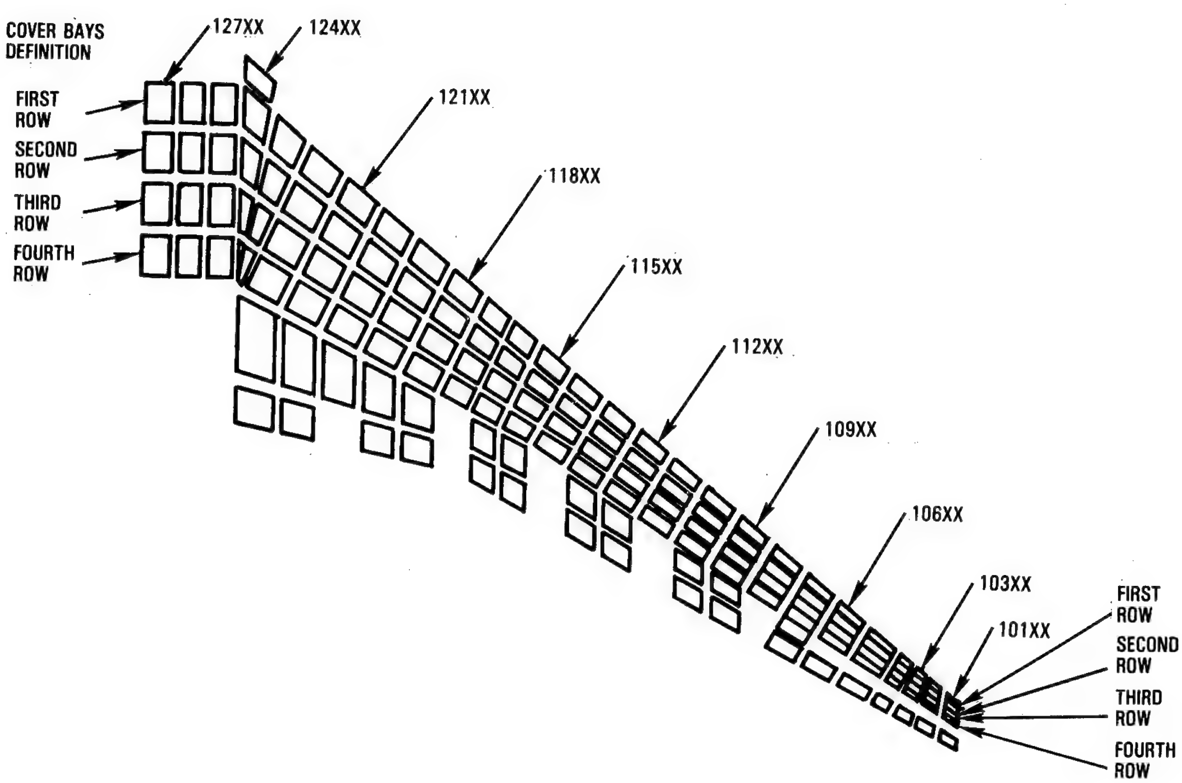


Figure 132. - Wing upper surface panel.

AIRCRAFT \ CONDITION	MANEUVER LOAD CONTROL APPLIED	MANEUVER LIMIT - 2 G FROM 2.5 G
REFERENCE AIRCRAFT AR 7.63 SWEEP 35°	-4.5%	
AR 12 SWEEP 35°	-5.1%	-11.7%

Figure 133. - Change in wing weight.

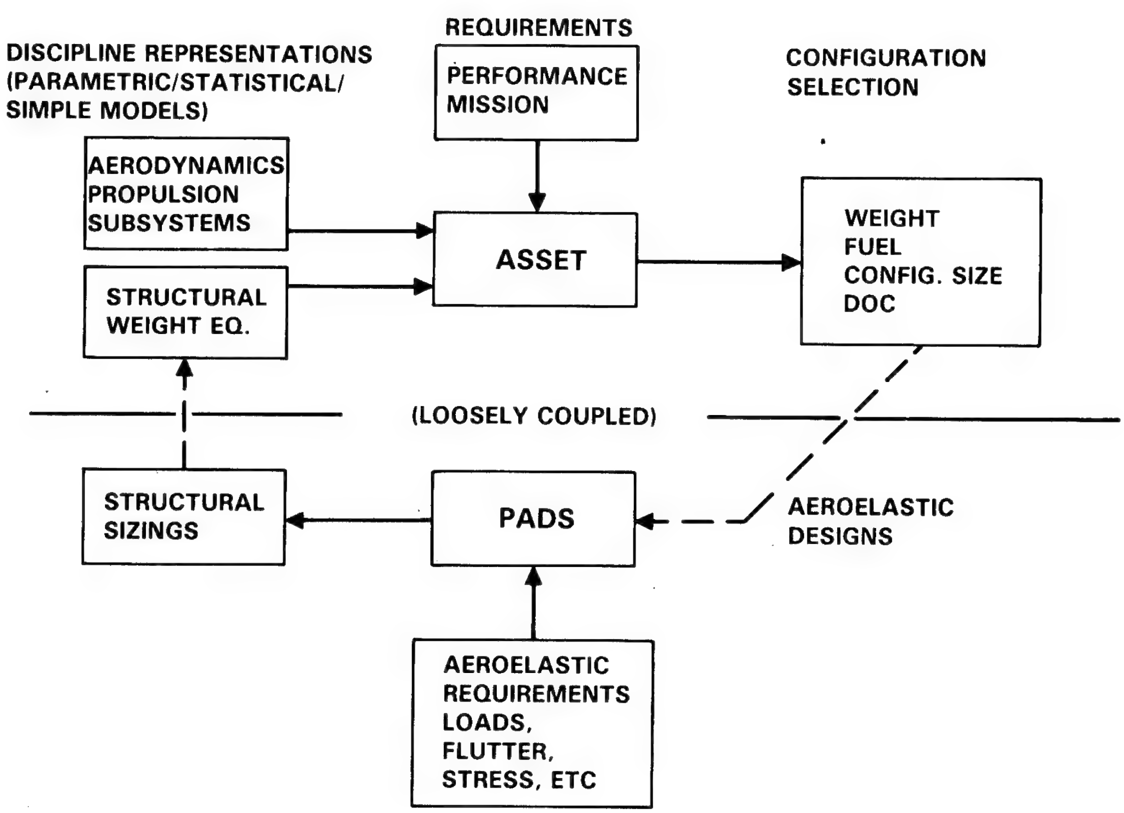


Figure 134. PADS and ASSET interface.

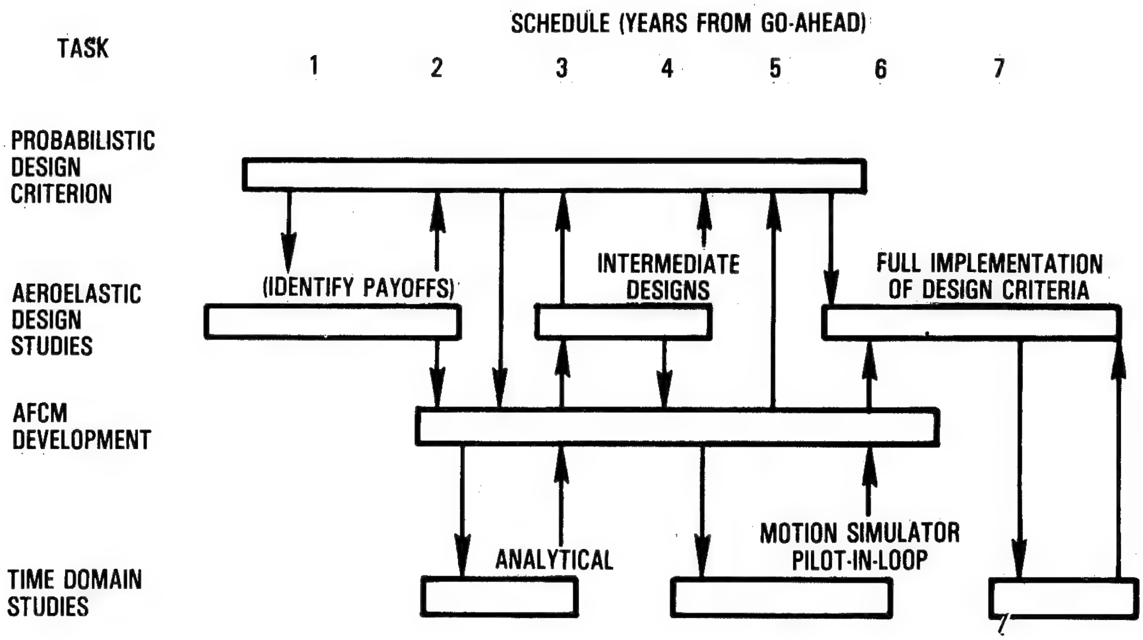


Figure 135. - Advanced flight control concepts development plan for alternate IDEA airplane.

10.5.1 Probabilistic design criterion definition. - The central issue to the feasibility of the full implementation of reduced design load factors is the establishing of probabilistic design maneuver criteria consistent with criteria associated with gust, taxi, landing, etc. This is a major undertaking.

10.5.2 Aeroelastic design studies. - The aeroelastic design studies will incorporate results and generate design data in support of the probabilistic design feasibility study. The primary concern of the aeroelastic study is to identify the areas which show the most promising structural weight reduction and performance increases. The design studies will define typical requirements for the definition of the AFCM function and will focus on the particular areas which will require simulation studies. The aeroelastic studies could proceed on a "what if" basis for a quick definition of the potential payoffs. The aeroelastic study will also identify the requirements associated with the advanced lead-type sensor development.

The study will establish the payoffs in wing weight due to reduced load factors, advanced airfoil technology, advanced planform geometries, and active control technology. The objective function may be minimum block fuel.

The proposed study will evaluate flutter requirements. The design loads will include gust effects if they are found to be critical. Payoffs for advanced active control technology will be identified relative to a known baseline aircraft and controls technology. Finally assessment of the potential payoffs will be summarized. Payoffs for reduced fuel burn will also be shown.

10.5.3 Definition of the AFCM. - The initial definition of the AFCM system will be dependent on the results of the Probabilistic Design study and the aeroelastic design study. The refined definition of the AFCM will involve the recycling of the AFCM results into the first two studies. The time deomain simulation studies will constrain the feasible design space of possible functions which may be implemented in the AFCM. The definition of the expert system will require extensive man-machine interface studies with primary data probably available through the Shuttle and advanced fighter programs.

10.5.4 Time domain study. - The function of the time domain study is to identify continually the time domain constraints on the definition of the AFCM and to verify the airplane performance in gust and wind shear using the advanced flight control concepts. A typical study may take the 2.5 g airplane and fly it with a standard flight control system with the pilot in the loop. A series of gust and/or shear inputs will cause high load factors and/or upset maneuvers. These same conditions will then be imposed on the airplane with the full authority flight control system to verify satisfactory performance of the system.

11. ALTERNATE IDEA ELECTRIC SYSTEM

11.1 Alternate IDEA Secondary Power

The Alternate IDEA, from a secondary power system standpoint, is impacted in this airplane primarily by the changeover from the engine driven fuel and lube pumps (in the IDEA) to electrically driven pumps. This change requires an upsizing in the individual generator capacities from 150/200 kVA to 220/275 kVA and a downsizing of the gearbox. The propulsion secondary power system in the Alternate IDEA therefore comprises the following:

- An "E^3+" technology engine
- An "austere +" AGB
- Two 220/275 kVA

The "austere +" gearbox is a further simplification of the auxiliary gearbox (AGB) design as a result of the elimination of the fuel/lube pumps and reduced accessory drive provisions. The size and torque-capacity of the power take-off shaft from the HP spool shaft, however, is not changed, since the power for the pumps (originally supplied mechanically) is now furnished by the generators in the Alternate IDEA.

As stated previously, it is possible to consider an integrated engine generator (IEG) design for the Alternate IDEA, in which the SmCo rotors of the two generators in each power plant are mounted directly over the high pressure spool shaft. This technology has been very well investigated and evaluated under AFWAL contracts by the General Electric Company and the feasibility has been established. Work has also been accomplished by Rolls Royce on the IEG. A typical configuration is shown in figure 136. Rolls Royce has a favorable perception of the IEG technology, but there is growing opinion that while the generator(s) within the engine would completely eliminate an AGB system and

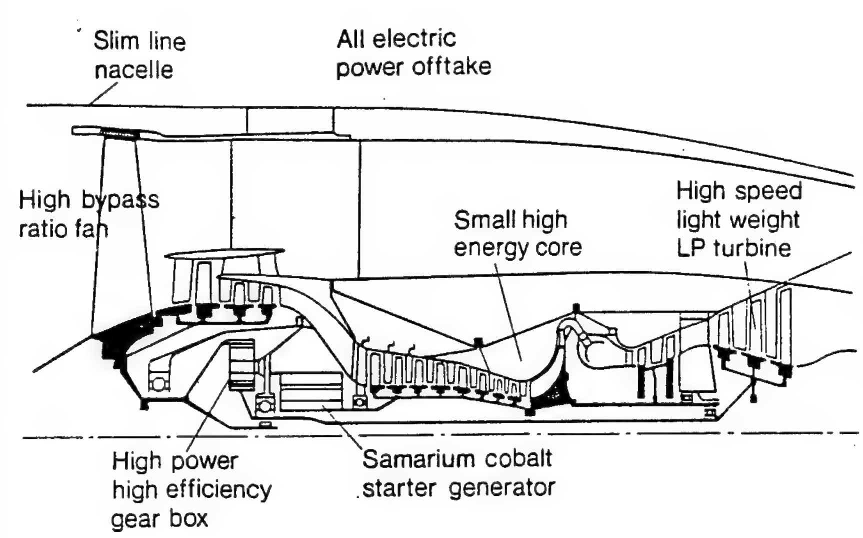


Figure 136. - Engine integrated starter generator (courtesy Rolls Royce).

its lubrication problems, the IEG concept lacks practical viability. Aside from the need to design the engine with a good deal of modularity to permit fast/easy removal of the bypass fan, there is the electrical consideration that the generator rotor speed would be restricted to that of the high pressure spool. Typically, the generator rpm could be much higher than the HP spool speed and this would also be an advantage, during engine starting, since the polar moment of inertia, referred to the S-G (starter-generator) would be much reduced. Because of these considerations, the Lockheed opinion also tends towards the use of an "austere + AGB", but it is hard to overlook the potential reliability that would come from a gearless-drive configuration for the IEG/S. Figure 137 shows a PTO shaft drive configuration on the GE CFM 56 engine design.

It will be possible in the Alternate IDEA to use the concept of tandem generators as described in the IDEA since, while these generator designs have all the appearances of a single machine from the mechanical packaging standpoint they are in reality two separate electric machines that take advantage of a single cooling loop. Also, by virtue of the integrated design concept it is possible in either the side-by-side or in-line configuration to synchro-phase the two machines and to parallel them electrically, without the problem of dynamic phase misalignments. This paralleling could be ofr advantage during starting since they could provide a stiffer power source as might be required for the tarting of large motors.

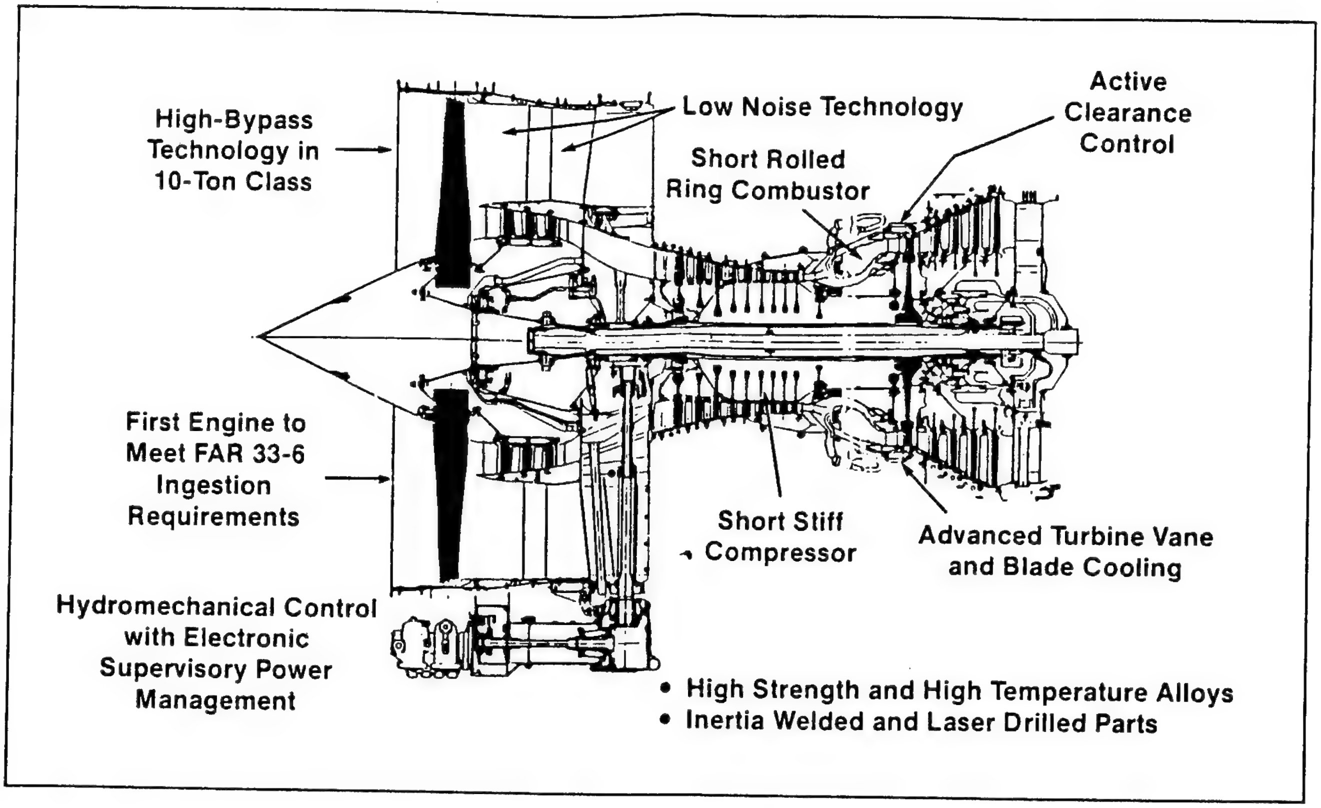


Figure 137. - High bypass turbofan with power takeoff accessory drive (courtesy General Electric).

11.2 Other Electrical Systems for the Alternate IDEA

In Chapter 10 it was stated that the proliferation of electronic and electric technologies was already making possible the application of advanced flight control that would permit more sophisticated interactions between the airframe, the propulsion and the flight control systems. This salutary relationship permits control of aerodynamically unstable aircraft and, in the case of commercial aircraft, the use of an automatic FMS and FADEC (full authority digital engine control) systems that will provide the most fuel-efficient operation of the airplane, as it fits into the advanced ATC systems.

The developments in VLSI and VHSIC, and the ever increasing improvements in packaging are the most significant enabling technologies in the foreseeable development of Alternate IDEA type aircraft. When the Alternate IDEA enters into airline service there will be a major inventory of advanced data sources, using laser, infrared, optical, and other technology. These technology

developments will permit the development of "lead-sensors" that could input data into the flight computers to control the airplane in any environment. Clearly, safety will in a key criterion in the Alternate IDEA, so lead-sensors could be used to detect wind shear, clear air turbulence (CAT) and gusts to permit control of the airplane as a stable platform in rough air conditions, etc. A by-product of technologies would be a significant improvement in ride-quality and improved all-weather landing capability. Other safety aspects relate to the adoption of sophisticated collision avoidance systems to provide a protective sphere around the airplane. This monitoring system would detect any penetration of the safety sphere by other aircraft and automatically initiate evasive action with respect to the computed speed, direction, attitude and altitude of the "other" aircraft. An early version of this type of system, known as Cockpit Display Traffic Information (CDTI), was flown successfully in a L-1011, but many improvements in CRT resolution and interpretation of its presented data are required.

There are many enabling electric/electronics technologies that will provide significant improvements in the safety and operational performance of future advanced aircraft in the time frame of the Alternate IDEA. Many of these associated with the sophisticated DFCS will permit wing structural load relief and consequent weight improvements. The development of high capacity electric generators/motors will also lead to improved capabilities and more innovative, diverse applications of electric machinery. Superconductivity and cryogenic cooling are without application in current aircraft, except in a few isolated cases. This technology and the use of exotic power sources will bring about major changes in the engineering development scenario of advanced aircraft.

Batteries, fuel cells, thermal batteries, new primary/secondary batteries and exotic batteries using consumable electrodes or liquid electrodes will make quantum jumps in energy level capabilities and specific outputs over the next twenty years. Where specific outputs are now in the range of 8 to 20 Wh/lb, the projected maximum theoretical possibilities of known exotic battery configurations (such as lithium sulphur, and lithium chloride can reach as high as 1200 Wh/lb. The successful development of such high specific output batteries will complement the continuing technology developments in magnetic material rare earth permanent magnets and new conductor/insulation systems.

Another fertile area for improvement will be in APU technology. Here it is clear that the use of powder-metallurgy and ceramics will permit higher turbine inlet temperature and higher specific output for less fuel. The all electric technology concept applied to the APU will bring about hardware simplification and performance improvements.

12. IMPACT OF IDEA TECHNOLOGIES

The impact of IDEA technologies on Direct Operating Cost (DOC) accrue from four major effects.

1. Improvement in engine performance resulting from bleed elimination.

2. Reduction in systems and other weight from incorporation of a digital flight control system and all-electric secondary power system.

3. Drag and weight reduction using a pitch active control system (PACS). The incorporation of a digital flight control system permits the use of PACS.

4. Improvement in maintainability and reliability.

The first three effects have an impact on block fuel savings with a corresponding reduction in aircraft empty weight as the aircraft is resized to achieve the design mission. The savings in block fuel and empty weight in turn produce a reduction in DOC. The fourth effect produces a direct saving in DOC.

The impact of IDEA technologies on block fuel is therefore the dominant effect in reduction of DOC and will be discussed in more detail.

When the IDEA technologies discussed in this report are incorporated into the aircraft, the ASSET performance and sizing program predicts 11.3 percent reduction in block fuel from the baseline configuration. The contributions to this reduction may be estimated using sensitivity factors established from previous studies. Sensitivity analysis provides a useful, although approximate, method of allocating block fuel benefits to the IDEA technologies. The sensitivity analysis is summarized in figure 138 in which the changes to the aircraft systems impact SFC, thrust, L/D, and OEW when the aircraft is sized for the design range of 4600 n.mi. and flown at the average stage length of 2500 n.mi. The block fuel impact is the product of the percentage change in these terms and the block fuel sensitivity factor. The net effect on block fuel for the aircraft is calculated by converting the percentage changes in block fuel into percentages of the baseline (e.g., a two percent reduction in block fuel converts to a block fuel that is 98 percent of the baseline). The total impact is found by taking the product of these percentages. This method is shown in figure 138 to agree closely with the ASSET prediction and will be used to show the impact of the components of the IDEA technologies.

12.1 Improvement in Engine Performance

As described in Chapter 8, in the IDEA configuration engine bleed is eliminated and the ECS is supplied with fresh air by an electrically driven compressor. The basic electric utility loads (e.g., galley and lighting) remain unchanged, as shown in figure 139. Actuator loads are slightly reduced because recirculation losses inherent in a hydraulic actuation system are eliminated.

	SFC	THRUST	L/D	OEW	BLOCK FUEL IMPACT
BLOCK FUEL SENSITIVITY FACTOR	1.4	−0.1	−1.4	1.15	
1. BLEED ELIMINATION	−1.5	+2.9			−2.4
2. WEIGHT REDUCTION					
— SYSTEMS				−2.8	−3.2
— OTHER				−0.6	−0.7
3. WEIGHT PENALTY					
— C.G. CONTROL				+0.5	+0.6
RSS — TAIL SIZE REDUCTION			+2.1	−0.6	−3.6
— TRIM DRAG REDUCTION			+1.9		−2.7
TOTAL FACTORED REDUCTION IN BLOCK FUEL					−11.5
ASSET PREDICTION					−11.3

Figure 138. - Estimation of block fuel reduction.

LOAD	BASELINE		IDEA	
BASIC ELECTRIC	196 kVA =	238 HP	196 kVA =	238 HP
ACTUATORS — HYDRAULIC		55 HP		
— ELECTRIC			35 kVA =	42 HP
ECS — BLEED	4.2 LB/S			
— ELECTRIC			232 kVA =	282 HP
TOTALS — BLEED	4.2 LB/S			
— HPX		293 HP		562 HP
PER ENGINE — BLEED	1.4			
— HPX		98		187 HP

Figure 139. - Aircraft power requirements at cruise.

The major difference in secondary power extraction between the Baseline and IDEA is bleed elimination and the additional electrical power requirements.

Figure 140 shows the impact of bleed and horsepower extraction on the E^3 engine at a typical operating condition of 85 percent maximum cruise power of 35,000 ft and M 0.8. The ECS compressors require about 66 hp for each 1 lb/sec of air so that the substitution of electrical power for bleed produces a net reduction in SFC penalty. Similarly (figure 141) there is also a reduction in net thrust loss i.e., a smaller engine may be used to generate the same thrust, although the effect on block fuel is small.

12.2 Systems Weight Reduction

Systems weight changes to the baseline aircraft before resizing to achieve the design mission are shown in figure 142. The weight changes are discussed in more detail in the following sections.

12.2.1 Flight controls. - The implementation of a digital fly-by-wire flight control system (DFCS) is described in Chapter 9. The incorporation of the DFCS produces weight savings in the autopilot, in mechanisms (through the elimination of cables, pulleys, rods, bellcranks, and their associated supports), and in multiplexing of electrical controls. The combined impact is a reduction of 819 pounds in flight controls weight, (figure 143).

In addition, there are weight savings from the substitution of hydraulic actuators with Electro-Mechanical Actuators (EMA). The major weight reduction is in the elimination of plumbing and fluid. The weight of power cables is in the accounting of weights under "electrical controls" but cable runs are short because power is carried only from the electric power bus to the EMA. The weights of the buses themselves appear under the electrical system group weights. EMAS produce a net weight saving of 585 pounds.

12.2.2 Hydraulic weight savings. - The plumbing and fluid weights shown in figure 143 apply only to hydraulic lines which are dedicated to a single flight control actuator. Lines which serve more than one actuator appear in the hydraulics weight group, and the remaining weight savings from the elimination of the hydraulic system are shown in figure 144. The major contribution to weight reduction is in plumbing and fluid weights.

12.2.3 Electrical system weight savings. - Weight differences are shown in figure 145.

The largest contribution to weight reduction results from multiplexing. The wiring weight subject to multiplexing is 1704 pounds; this value includes wire weights which appear in other weight groups such as flight controls. Multiplexing reduces these wire weights by 1114 pounds, although there is an offsetting increase of 266 pounds required for electrical power wiring, giving a net weight reduction in wiring of 848 pounds. Of this value 682 pounds is attributed to the electrical system weight group. The balance of 166 pounds appears in other weight groups.

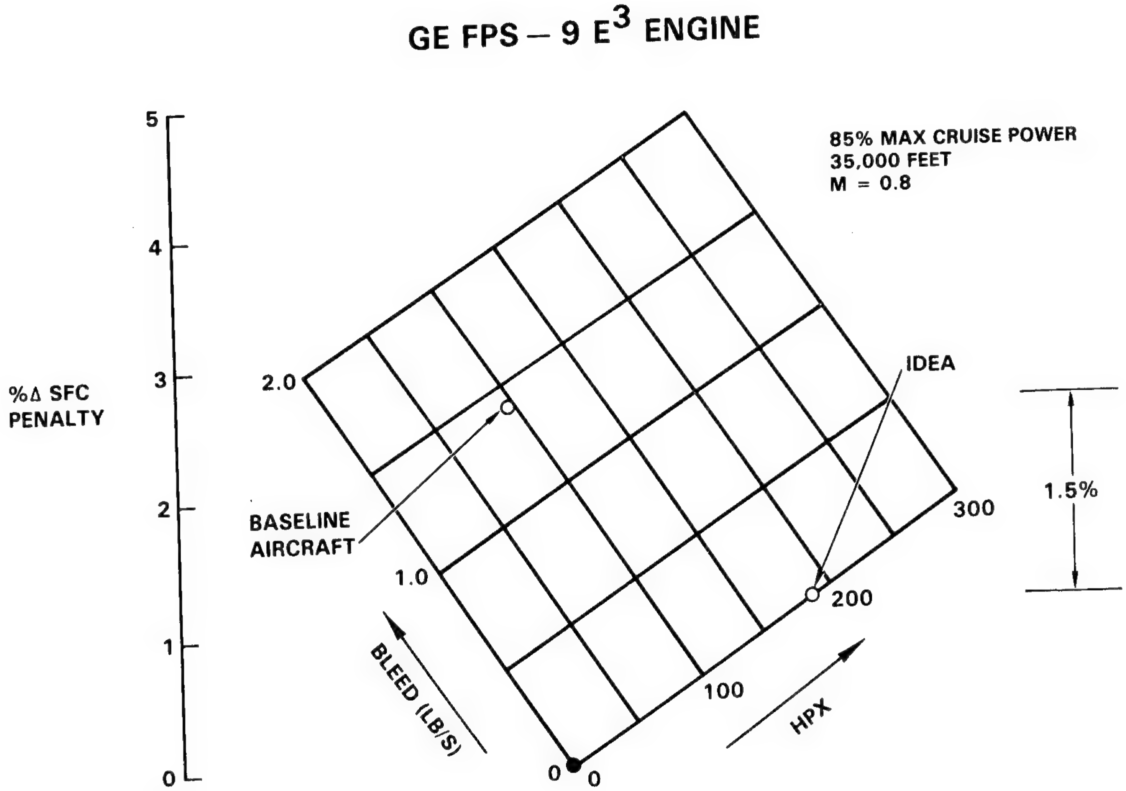


Figure 140. - Effect of bleed and horsepower extraction on SFC.

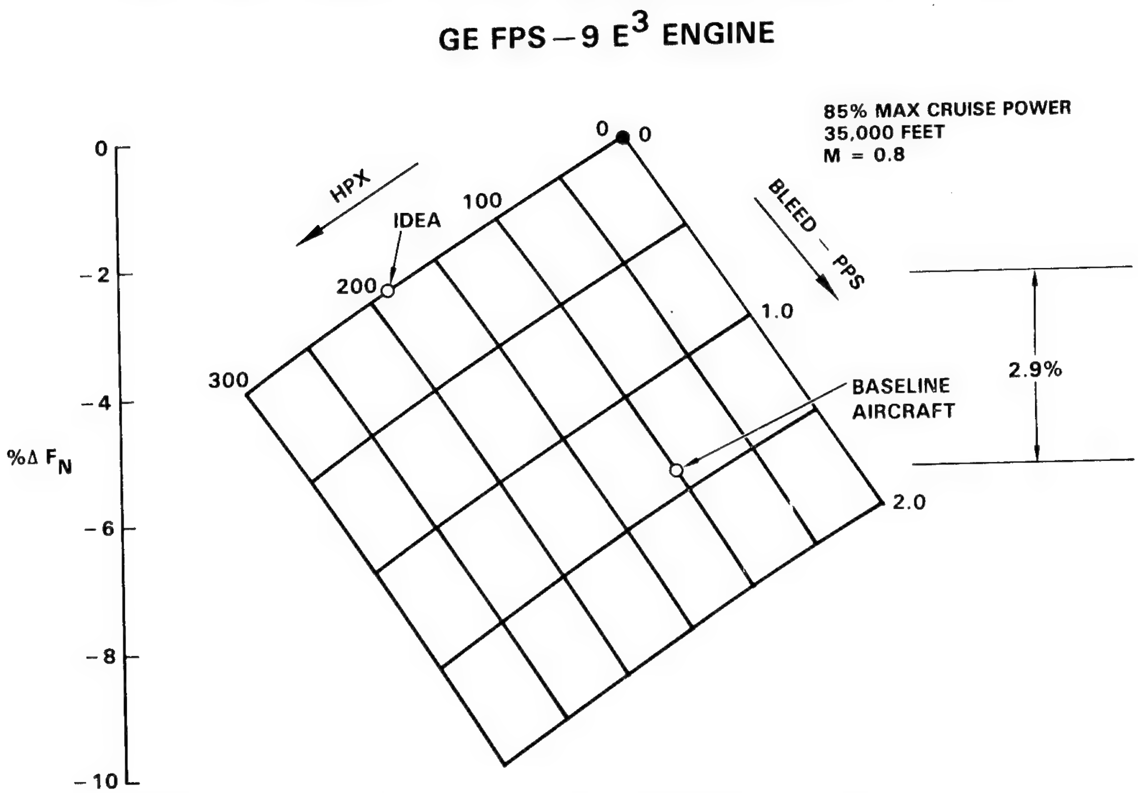


Figure 141. - Effect of bleed and horsepower extraction on maximum cruise thrust.

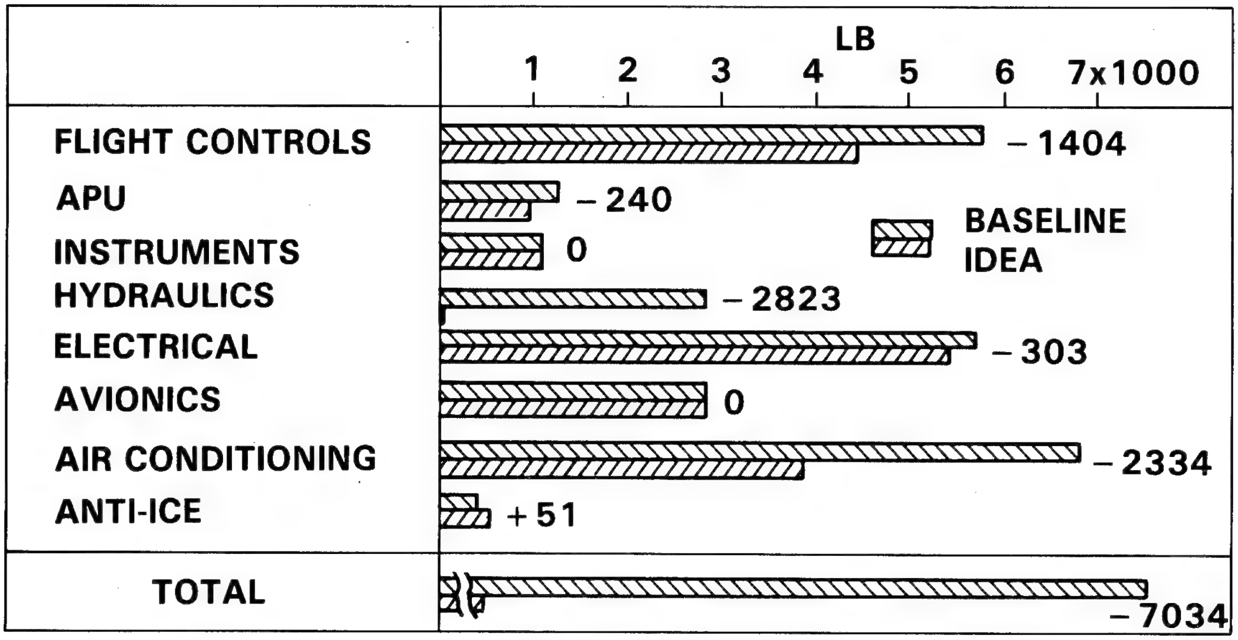


Figure 142. - Systems weights and savings.

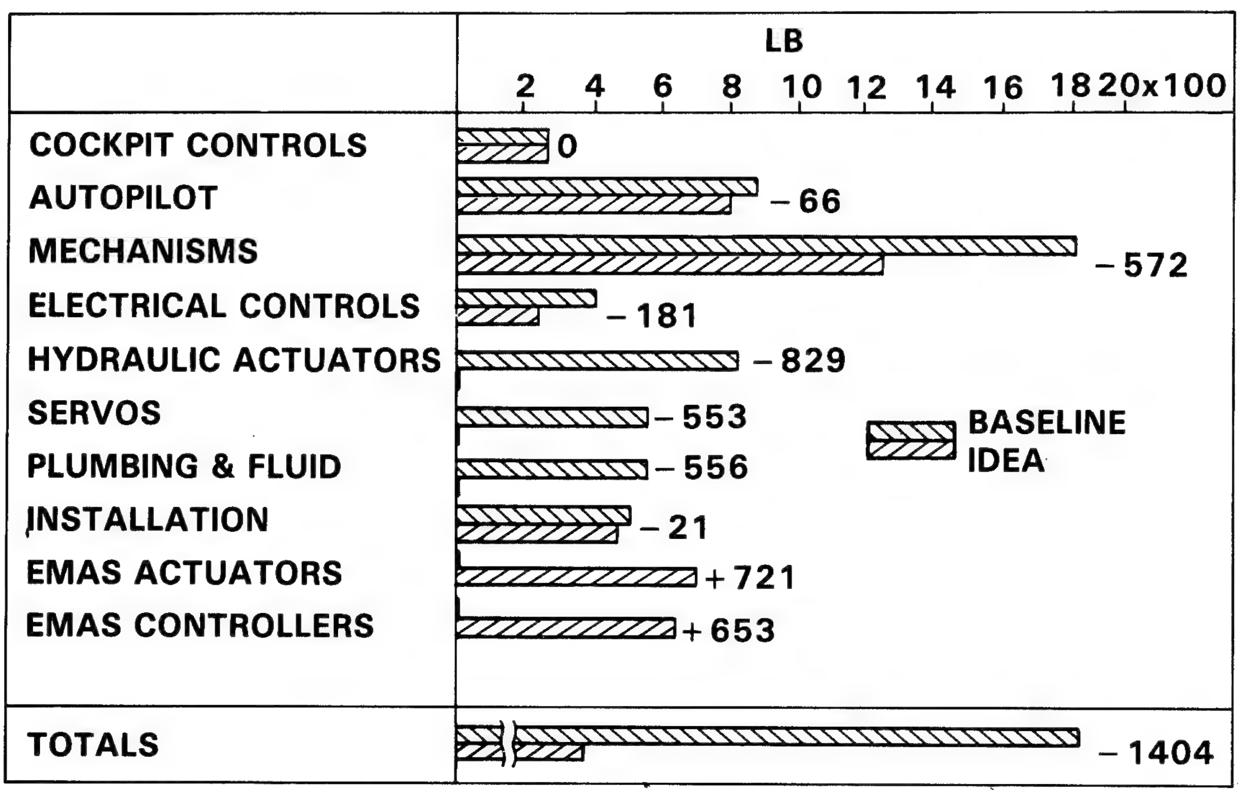


Figure 143. - Flight controls weight savings.

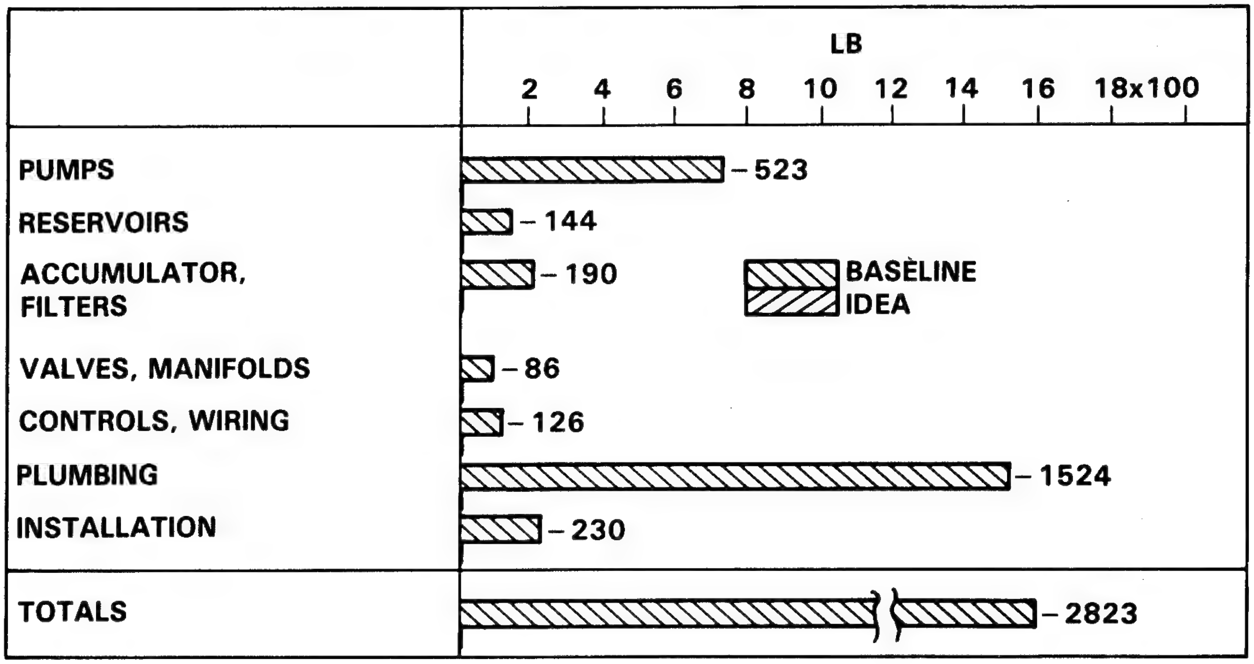


Figure 144. – Hydraulics weight savings.

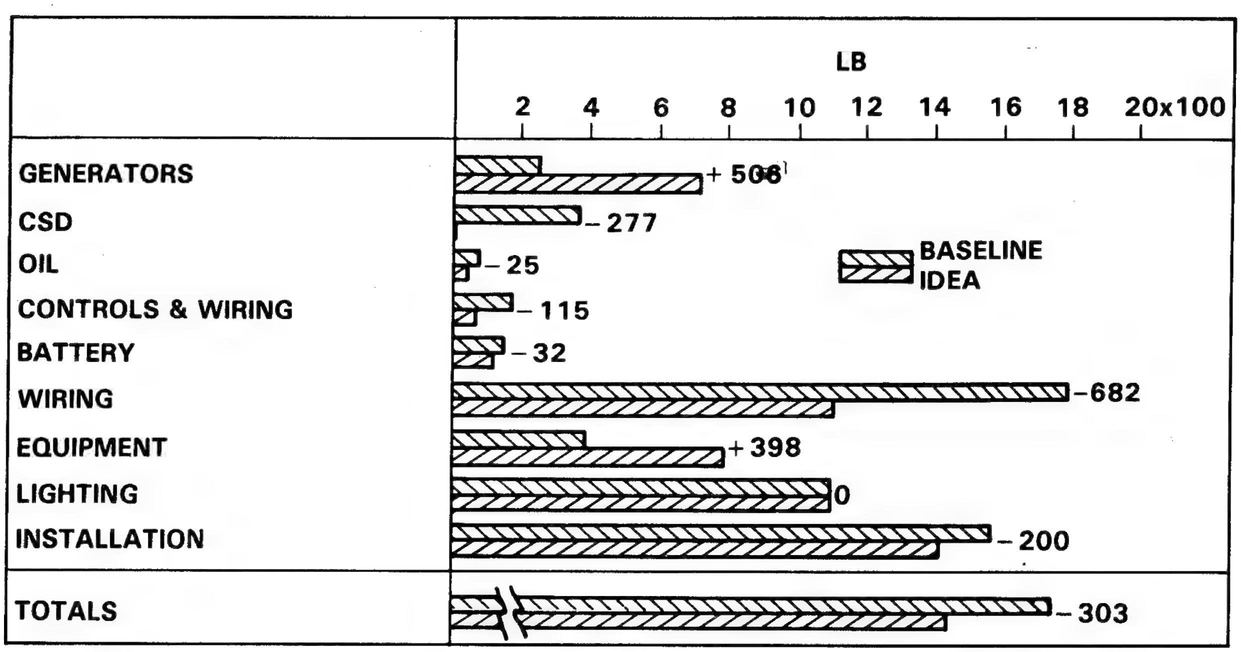


Figure 145. – Electrical system weight savings.

Instruments and their associated avionics were considered outside the terms of reference of the study, and their wire weights were therefore not included as subject to multiplexing weight reduction. Had these been included in the study, it is estimated an additional 500 pounds could have been saved.

12.2.4 ECS weight savings. - Weight savings are shown in figure 146. The weight savings occur in the elimination of bleed air ducting from the engines to the ECS packs and elimination of the bleed air precoolers and controls.

12.2.5 Other weight savings. - In addition to the weight savings described above, there are also weight savings in the engine and landing gear (figure 147). Elimination of the pneumatic engine starter saves 132 lb/engine. Elimination of the starter and CSD drive pads from the accessory gearbox permits a more austere gearbox to be used; however, there are now two electrical generators/engine, so only one pad is eliminated. Additional weight savings can be achieved with the substitution of the thrust reverser air motors with electrically driven actuators.

12.2.6 Net weight reduction. - Net weight savings in systems, propulsion, and landing gear amount to 8492 pounds, equivalent to 3.4 percent reduction in OEW. Of this weight reduction 819 pounds is attributed to flight control systems weights savings resulting from incorporation of the digital FCS. The balance of the weight reduction results from the all-electric secondary power concepts. The impact of these weight savings on block fuel is about 3.9 percent.

12.3 Relaxed Static Stability

12.3.1 Description. - The digital fly-by-wire flight control system incorporates a pitch active control system which permits the aircraft to be balanced with a negative static margin. This results in the benefits to aircraft performance resulting from the following effects:

1. A reduction in tail area reduces skin friction drag.

2. Reduced tail area saves structural and actuator weights.

3. Reduced tail download reduces wing lift requirement and hence induced drag.

With the same c.g. range as the baseline configuration, the horizontal tail volume coefficient can be reduced to $V_H = 0.62$. This requires a design with sufficient redundancy in the augmentation system to ensure that the system will not fail. An inadequacy with this approach is that the criteria to define the control limits of this aircraft are currently insufficient. A suitable definition of these control limits for a particular airplane requires a thorough definition of the aerodynamic characteristics (including any high-angle-of-attack nonlinearities) and control system networks to be used as a mathematical model for a detailed dynamic analysis. This model must then be exercised in various levels of turbulence, discrete gusts, and critical flight maneuvers to determine combined augmentation and control requirements which

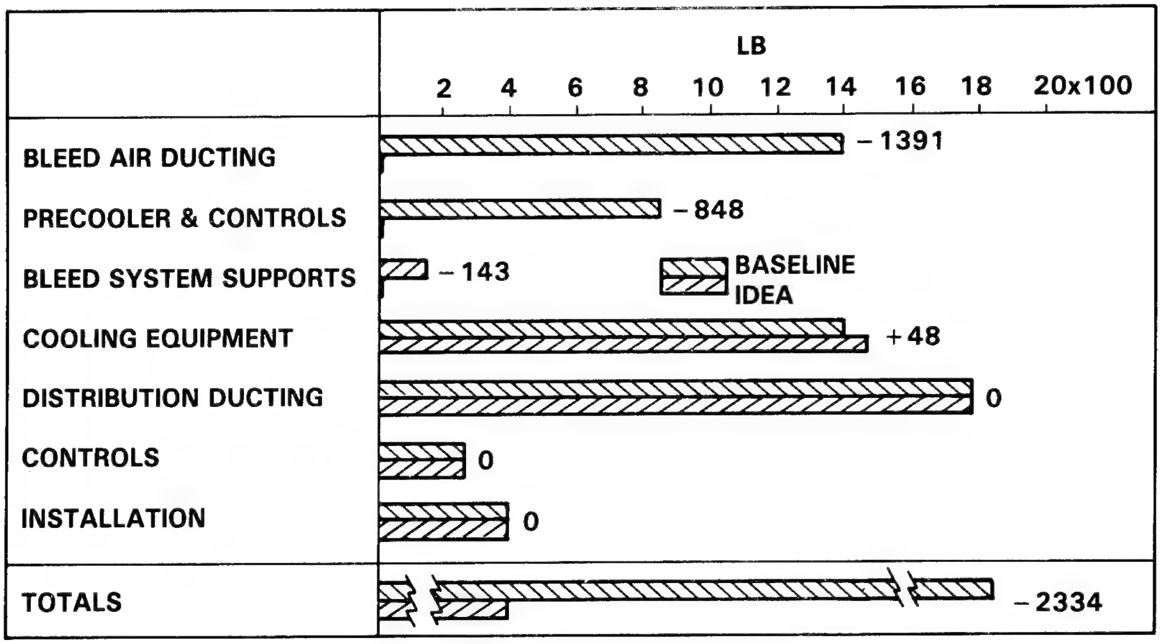


Figure 146. - Environmental control system weight system.

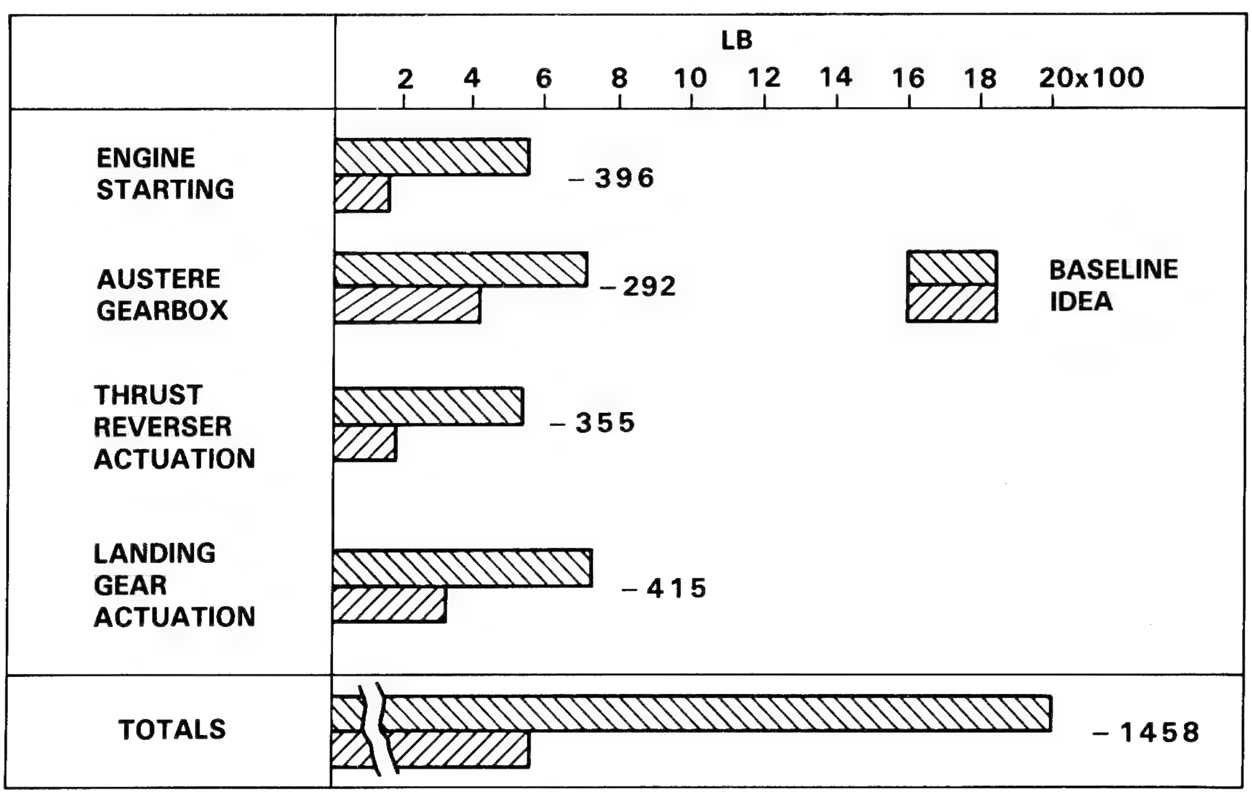


Figure 147. - Other weight savings.

dictate the horizontal tail size and/or maximum aft c.g. location for that particular design. The depth of this analysis is beyond the scope of the current study. However, it is sufficient to point out that the tail sizes obtained in this study meet the nose-down control requirements at aft c.g. for the required nose-down control angular acceleration at stall recovery.

For this study the horizontal tail was sized in accordance with the following requirements:

- Total c.g. range of 57.6 inches
- Takeoff control-to-stall at forward c.g. with maximum takeoff flaps at 1-g stall speed
- Nose-down angular acceleration of -0.09 rad/sec for stall recovery

The horizontal tail was assumed to be an all-moving surface with 30 percent chord geared elevator yielding a usable $C_{L_{max}}$ of -0.9 at a -14 degree stabilizer deflection. It is possible that a further reduction in tail size can also be achieved by assuming a higher tail lift curve slope. This would entail an expanded study including different tail planforms. Such an analysis is beyond the scope of this study.

Downwash data at the tail in reference 25 were modified slightly for tail arm length differences.

Figure 148 is a tail sizing notch chart which compares the tail size requirements for the baseline and IDEA configuration. In both configurations the forward limit of c.g. travel is constrained by control to stall at landing configuration. For the baseline configuration the aft limit of c.g. travel is limited by the requirement for a six percent static margin from the neutral point locus.

For the IDEA configuration the PACS allows the aft c.g. limit to have a negative static margin. A system has been developed at Lockheed for a -10 percent static margin and has been demonstrated in piloted simulation at NASA Langley (ref. 26). This value was therefore selected for the IDEA configuration, although a more negative margin may be feasible.

Note that the notch chart is premised on the c.g. range as a fraction of MAC before resizing. The c.g. range for the resized wing, as a fraction of MAC, would increase by 2.7 percent because of a corresponding decrease in MAC. This would require a slightly larger tail volume coefficient. Within the time constraints of this study this iteration could not be performed but the resultant error is small.

An additional constraint to the aft c.g. is the ability to maintain tail uplift. Reoptimization of the horizontal tail geometry would move this constraint line to the right permitting a slightly smaller tail volume. However, an additional constraint on tail volume, not defined in this study, is that a significant portion of the trapezoidal tail area is contained within the aircraft fuselage. This limits the effectiveness of the horizontal tail.

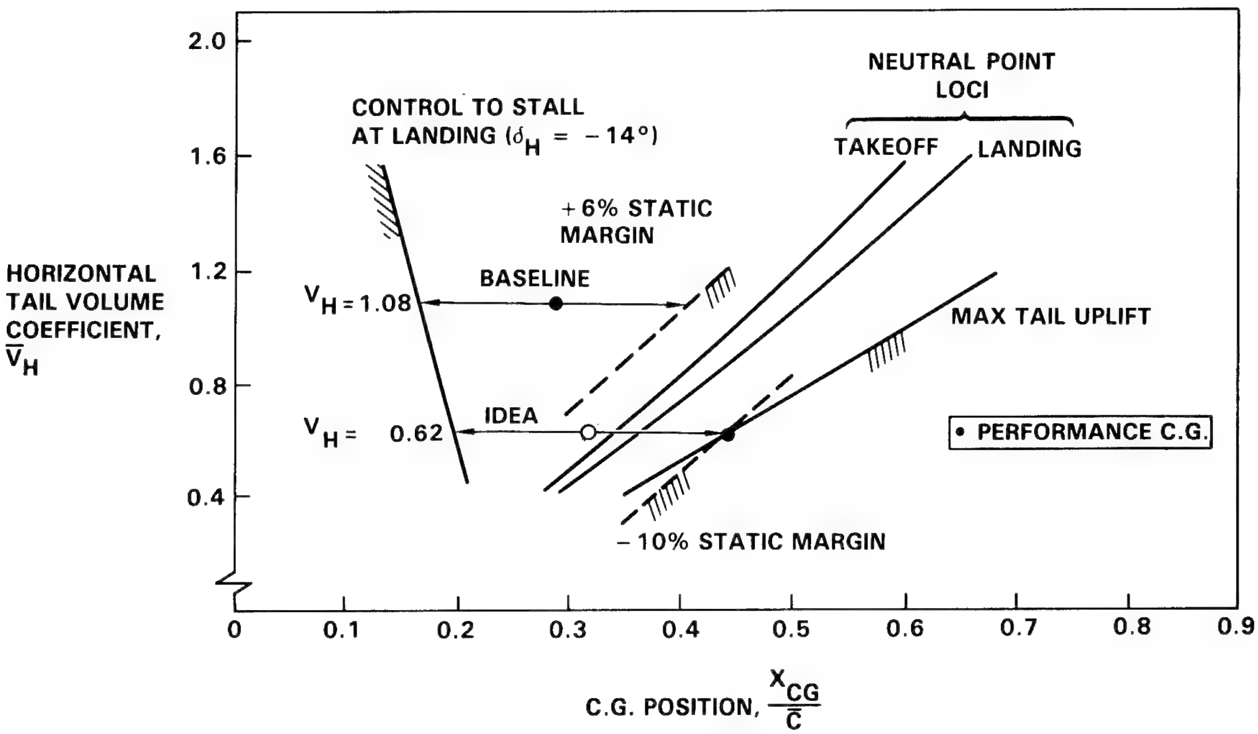


Figure 148. - Horizontal tail sizing.

An additional performance improvement is obtained using c.g. management. A saddle tank is located above the S-duct of the center engine with a capacity of 10,500 pounds of fuel. In addition, the vertical stabilizer is designed as a wet tank with capacity for an additional 10,500 pounds (figure 149). During climb and cruise, fuel may be transferred into the trim tanks to move the aircraft c.g. to its aft limit. Figure 150 shows that for a limited area of the c.g. envelope, the trim tank capacity is insufficient to move the c.g. to 44 percent MAC. These combinations of weight and c.g. would occur very infrequently, so that a c.g. for performance calculations of 44 percent can be assumed.

A similar system to the above is planned for the Airbus Industrie A310-300 in which 11,000 pounds of fuel may be transferred to tanks in the horizontal tail to move the c.g. to the aft limit (ref. 27).

12.3.2 Impact on aircraft weight. - The trim tank and its associated pumps, controls, and fuel piping is estimated to weigh 1168 pounds. This represents about one-half percent of OEW. This is more than offset by a reduction in horizontal tail weight of 1630 pounds before resizing the aircraft, resulting from a reduction in tail area.

12.3.3 Impact on aircraft drag. - Horizontal tail wetted area is reduced from 720 ft^2 in the baseline aircraft to 374 ft^2 on the IDEA configuration. This produces a reduction in skin friction drag which is 2.1 percent of total aircraft drag.

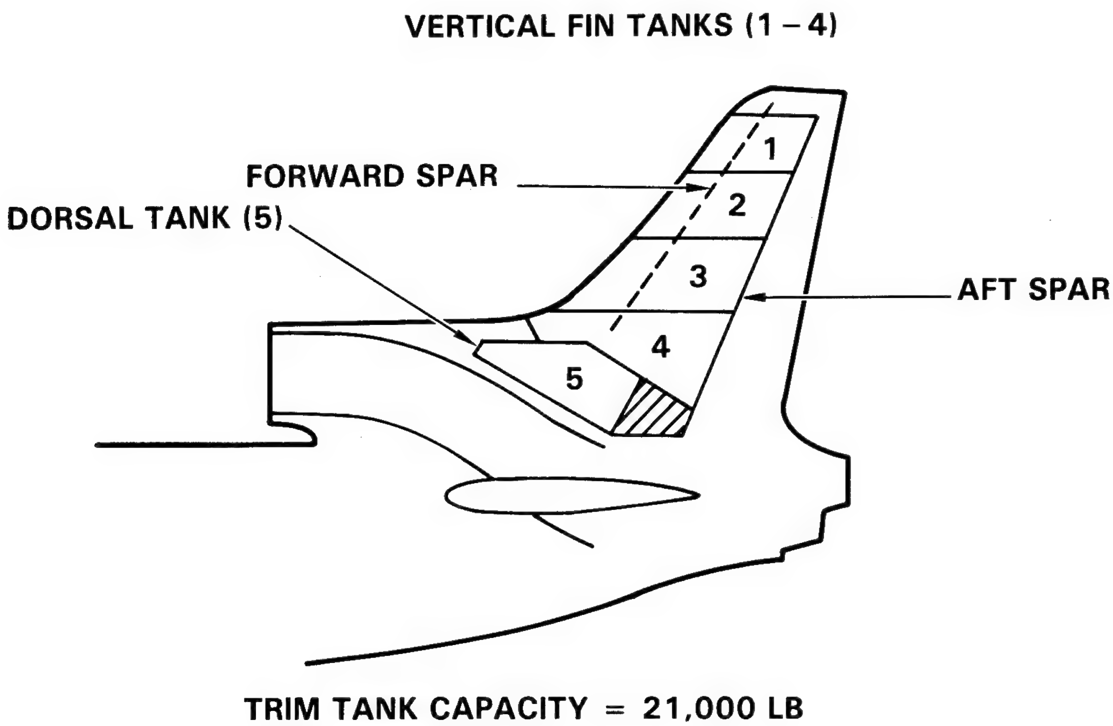


Figure 149. - Trim tank locations.

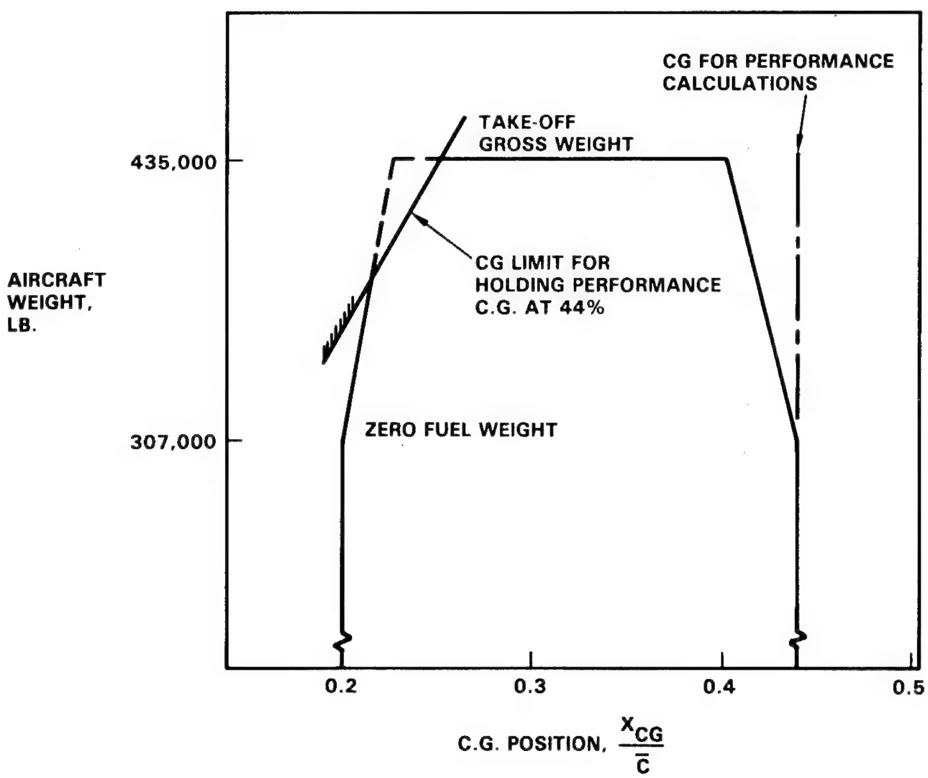


Figure 150. - C.g. transfer capability.

Aft movement of the c.g. produces a reduction in tail download, and hence a decrease in wing lift and induced drag. Relaxation of the static stability margin results in a 3 percent MAC shift aftward in the c.g. range. This produces 0.4 percent increase in L/D at a C_L of 0.5 (figure 151). The greater benefit occurs as a result of c.g. management, in that the c.g. for performance calculations is shifted from 32 percent to 44 percent MAC, producing an additional 1.5 percent improvement in L/D.

12.4 Reduction in Maintenance Costs

In the NASA/Lockheed study of ref. 2, reductions in maintenance costs were estimated for the IDEA concepts. These are shown in table 31 to reduce overall maintenance by 7.0 percent at the average stage length. Maintenance represents 19 percent of DOC so that these maintenance cost savings represent 1.3 percent reduction in DOC.

12.5 Reduction in Direct Operating Cost

Differences in weights between the baseline and IDEA configurations are shown in figure 152. Maximum takeoff gross weight is reduced 7.5 percent for the IDEA configuration and OEW is reduced 8 percent.

The first costs of the digital flight control and all-electric systems are not significantly different from the current technology systems which they replace, so that the cost/lb OEW is approximately the same for both baseline and IDEA aircraft.

Differences in DOC between the baseline and IDEA concepts are shown in figure 153. Fuel and oil represents 45 percent of DOC so that the 11.3 percent reduction in block predicted by the ASSET program contributes 5.1 percent reduction in DOC. The depreciation term in DOC is proportional to purchase price. As mentioned above, the cost/pound OEW is approximately the same for both configurations, and since depreciation represents 21 percent of DOC, the 8 percent reduction in OEW produces 1.7 percent reduction in DOC. The remaining contribution to the reduction in DOC is 1.3 percent reduction in maintenance costs. The net effect is to reduce the DOC of the IDEA configuration by 7.9 percent as compared with the baseline.

12.6 Differences in Aircraft Geometry

The major external differences in aircraft geometry are illustrated in figure 154 and described as follows:

12.6.1 <u>Reduction in wing area</u>. The wing loading is held constant at 131 lb/ft^2. This value approximates the nominal constraint for maintaining a constant value of fuel volume available to volume required of 1.1 (In practice, this value varied slightly from 1.09 for the baseline configuration to 1.08 for the IDEA configuration. Wing area is therefore reduced proportionally to TOGW).

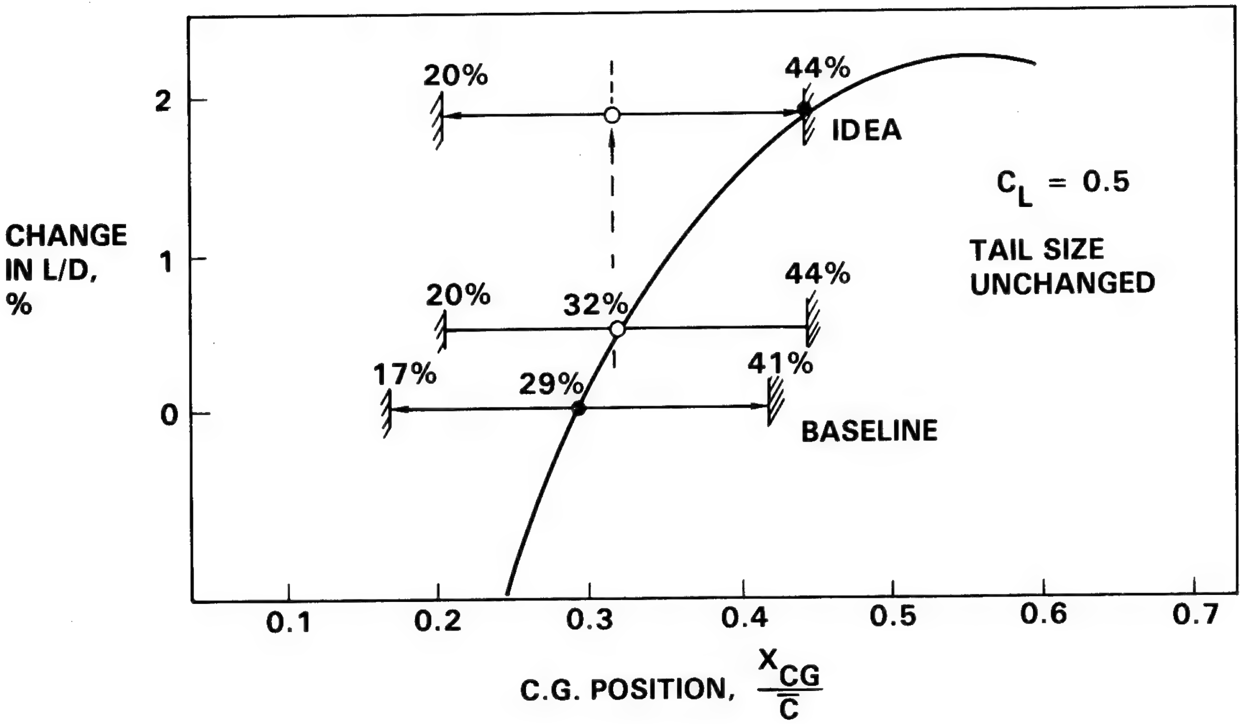


Figure 151. - IDEA trim drag reduction.

TABLE 31. - REDUCTION IN MAINTENANCE AS PERCENTAGE OF BASELINE

	Labor		Material		Reduction in Total Maintenance
	Per Cycle	Per Hour	Per Cycle	Per Hour	
All-Electric Secondary Power	-3.6%	-1.2%	-0.2%	-0.4%	-1.2%
Digital Flight Controls	-6.2%	-8.6%	-1.2%	-3.6%	-5.7%
Totals	-10.3%	-10.0%	-1.4%	-4.0%	-7.0%

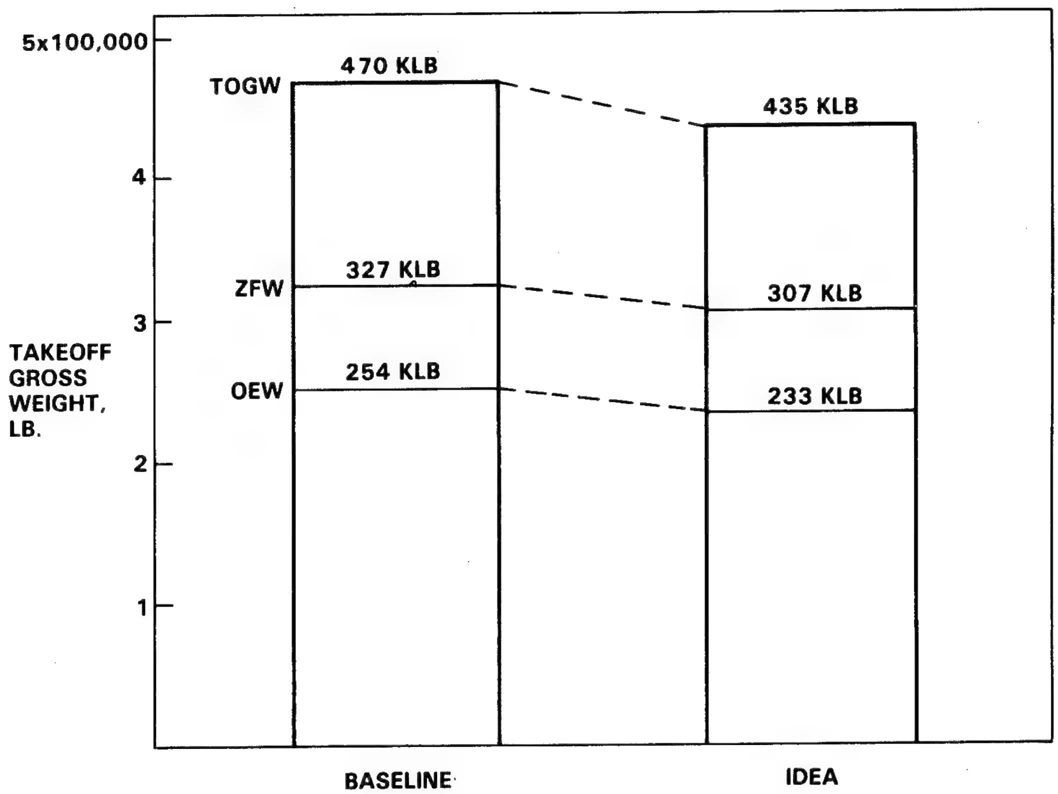


Figure 152. - 7.5% reduction in takeoff gross weight.

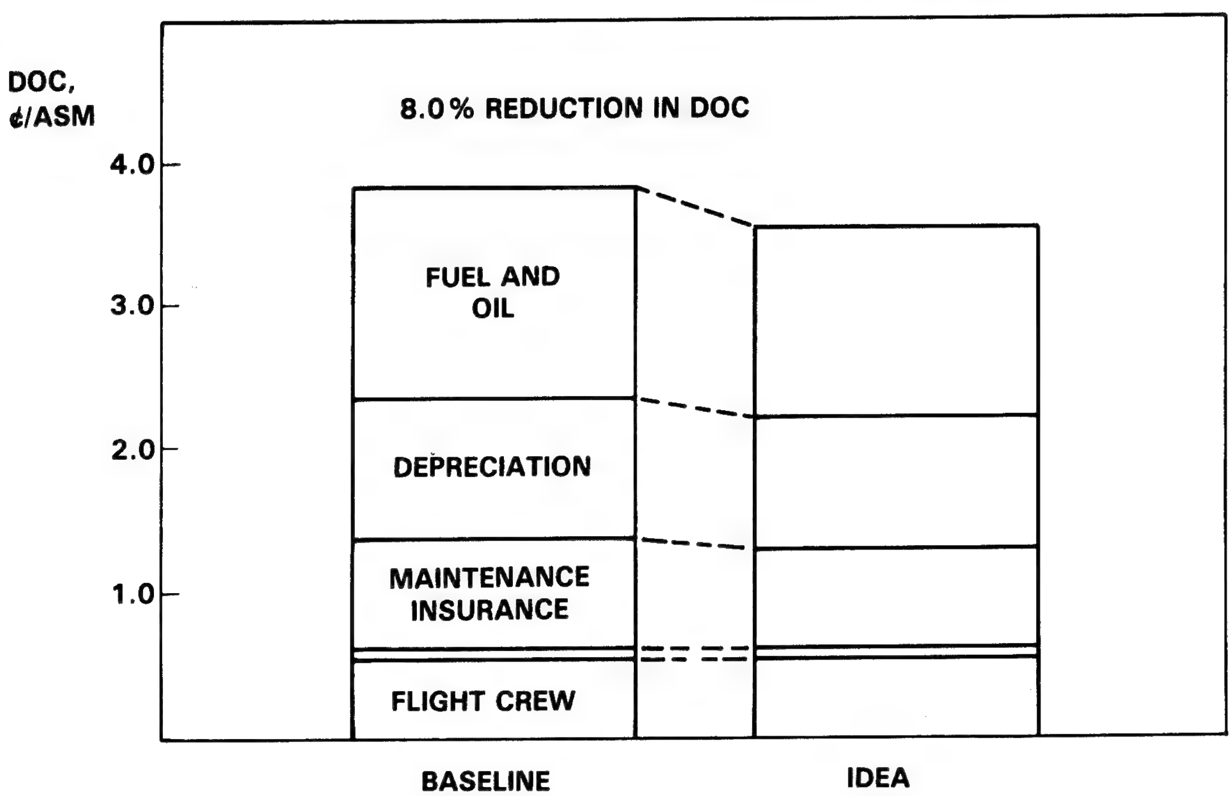


Figure 153. - Elements of DOC.

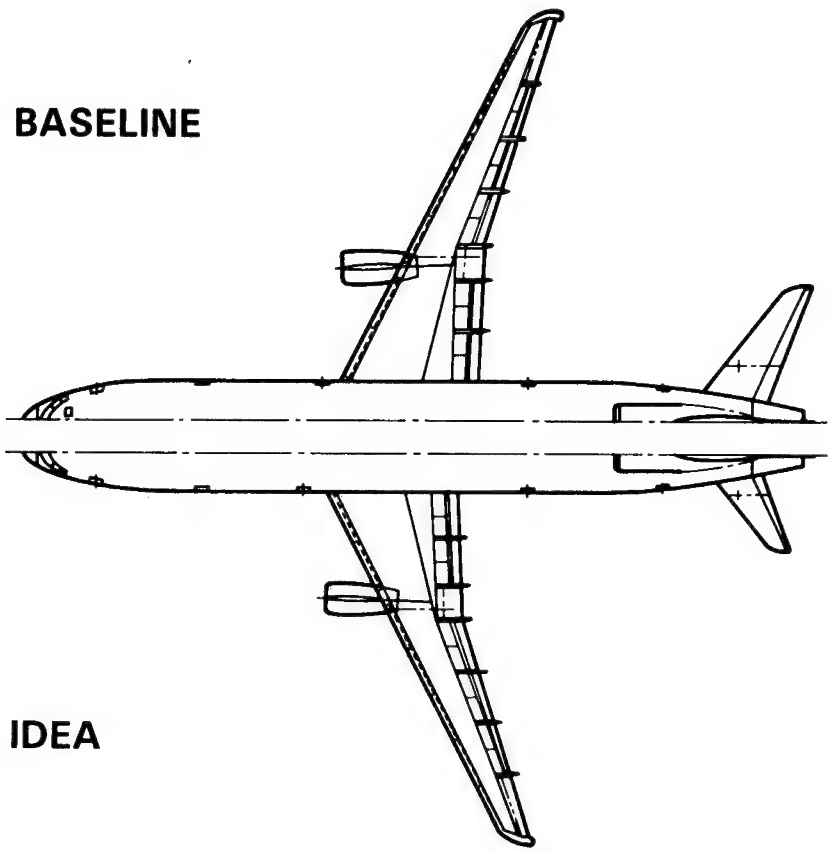


Figure 154. - Planform comparison.

12.6.2 Relocation of the wing. - As described in Section 12.3, the c.g. range of the aircraft is moved aft by 3 percent on the MAC, or 0.56 foot. Wing, fuel, engine, and landing gear weights represent 46 percent of TOGW so that the fuselage must move aft by 1.1 feet relative to the wing in order to move the c.g. range by 0.56 foot.

Rearward shift of the c.g. range requires that the location of the main landing gear be moved aft along its buttline chord. For an aft c.g. on the MAC of 44 percent, the landing gear post must be installed at 75 percent of the MLG buttline chord. This value is very close to the value of 74 percent established in ref. 4 as the limit for a landing gear design of the L-1011 type using a vertical landing gear post; no penalty in landing gear weight was taken.

12.6.3 Horizontal tail size reduction. - The reduction in horizontal tail volume coefficient from 1.08 to 0.62 results in a corresponding reduction in tail trapezoidal area.

13. IMPACT OF ALTERNATE IDEA TECHNOLOGIES

The Alternate IDEA technologies evaluated in this chapter are primarily those associated with the adoption of an advanced flight control manager, which would permit a reduction in the design load factor, as discussed in Chapter 10. In addition, some additional system weight reductions were made. These systems are:

- Landing Gear
- ECS
- EMAS
- Power Generation

13.1 Reduction in Design Load Factor

Figure 155 shows the effect of a reduction in wing load factor on block fuel when the IDEA configuration is designed for a range of 4600 n.mi. and flown at the average stage length 2500 n.mi. At an aspect ratio of 12, a block fuel reduction of 3 percent is obtained for a 2 g wing design and 6.1 percent for a 1.5 g wing design.

For a given design load factor, increasing aspect ratio to reduce block fuel results in a heavier wing and more expensive aircraft. The aspect ratio for minimum DOC therefore is lower than for minimum block fuel. These are plotted in figure 156 for a fuel price of $1.50/gal. The aspect ratio of the baseline configuration was optimized at a fuel price of $2.12/gal, which was the price assumed at the start of the study. At the request of NASA the price was reduced to $1.50/gal in order to be consistent with the Boeing Company, the other contractor in this study. At a fuel price of $2.12/gal the unconstrained optimum aspect ratio is 13, but the design constraint of installing the landing gear constrains aspect ratio to 12. For evaluation of IDEA concepts, nonoptimum aspect ratio does not affect results. It is only when considering aspect ratio changes that the choice of baseline aspect ratio becomes significant.

The curves of figures 155 and 156 do not include the penalties associated with installing landing gear in high aspect ratio wings. As aspect ratio increases the MAC moves aft relative to the buttline chord of the MLG. Above an aspect ratio of about 12, weight penalties become significant that are associated with installing the landing gear in the wing while providing a sufficient tip-up margin at the aft c.g. limit.

For the 2 g wing design wing weight is reduced by 13 percent and for the 1.5 g design a 28 percent reduction in wing weight could be achieved.

13.2 Reduction in Landing Sink Rate

Consistent with the methods of aircraft control described in Chapter 10, it is anticipated that the advanced flight control systems features on the

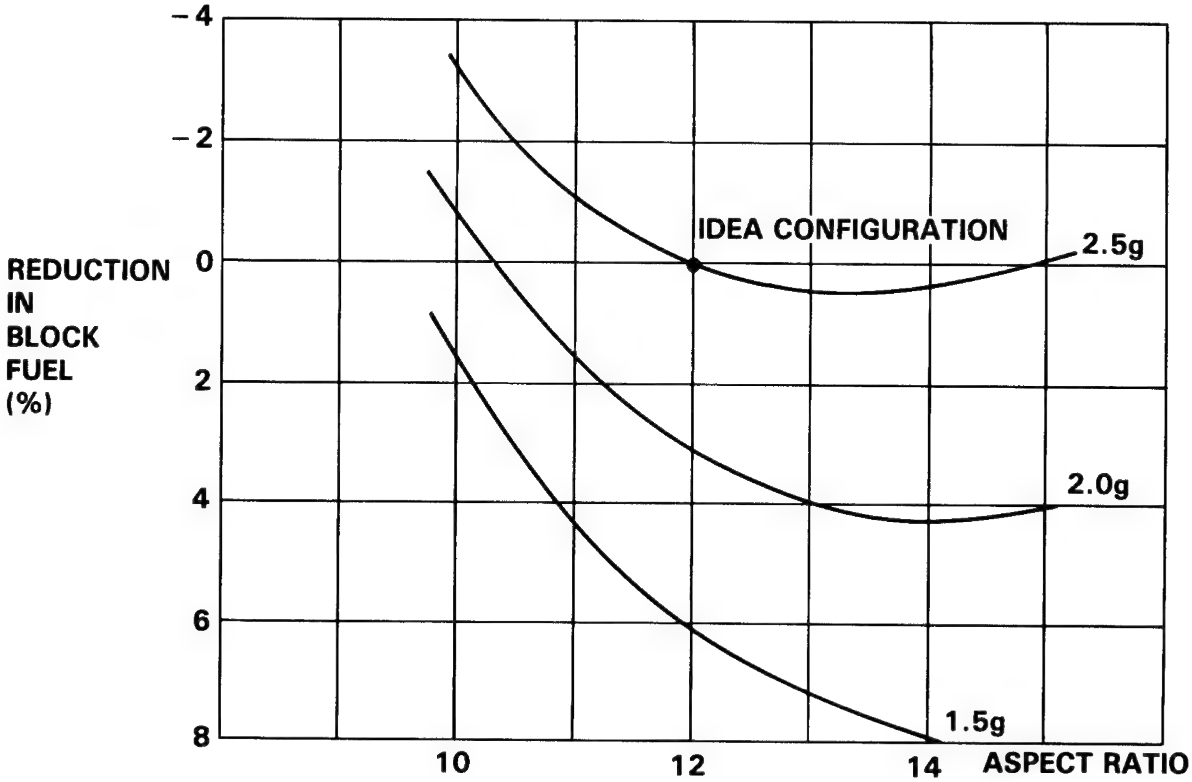


Figure 155 - Effect of design load factor on block fuel.

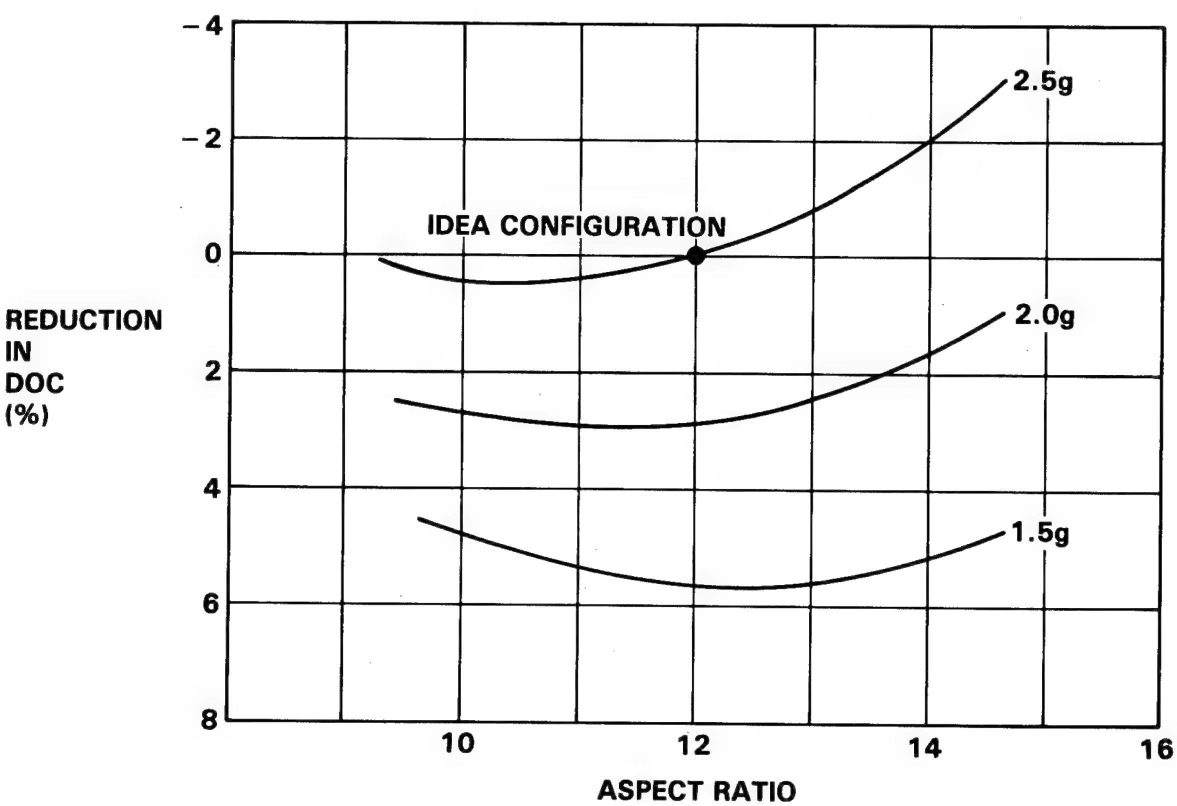


Figure 156. - Effect of design load factor on direct operating cost.

Alternate IDEA configurations will permit a reduced landing sink rate. Although no supporting analysis has been performed, the assumption has been made that the design sink rate may be reduced from 10 ft/sec to 5 ft/sec. It is estimated that landing gear weight could be reduced by 5.1 percent, or 999 pounds before the aircraft is resized for the mission. Some additional weight reduction would be achieved from wing structural components that are sized by landing loads.

13.3 Additional Reductions in Subsystems Weights

Further improvements to the subsystems in the IDEA configuration that would be available for the Alternate IDEA configuration are described more fully in Chapter 11. These weight reductions, along with the reductions in wing and landing gear weight, are listed in table 32.

13.4 Reduction in DOC

The combined effects of the Alternate IDEA concepts produce a total reduction in block fuel of 3.4 percent when the aircraft is resized. This reduction is achieved only as a result of airframe and systems weight reductions; there are no differences in L/D or SFC between the IDEA or Alternate IDEA concepts.

14. CONCLUSIONS AND RECOMMENDATIONS

14.1 IDEA Benefits

14.1.1 Block fuel reduction. - The impact of IDEA concepts on block fuel is summarized in figure 157. When the SFC, weight, and drag reductions associated with the implementation of IDEA concepts are applied to the 350 passenger Baseline Aircraft, the ASSET performance prediction program shows an 11.3 percent reduction in block fuel when the aircraft is sized for the design range of 4600 n.mi. and flown at the average stage length of 2500 n.mi. One-half of this benefit is attributed to all-electric secondary power through bleed elimination and systems weight reduction. The other half of the block fuel reduction results from the implementation of the advanced digital fly-by-wire flight control system which results in additional weight savings and permits a flight-critical pitch active control system.

14.1.2 DOC reduction. - IDEA concepts produce an eight percent reduction in DOC, figure 158. This reduction is achieved as a result of the reduction in block fuel and the associated downsizing of the aircraft, together with a small contribution to reduction in DOC from improvements in maintenance costs.

TABLE 32. - ALTERNATE IDEA WEIGHT SAVINGS

	Weight * Saving (lb)
2g Design Load Factor	6590
5 FPS Landing Gear	999
ECS Weight Reduction	2991
Advanced EMAS	72
Advanced Power Generation	98
Total Weight Saving	8,050
Block Fuel Reduction	3.4%
DOC Reduction	3.1%

*Before Resizing for Mission

14.2 Alternate IDEA Benefits

14.2.1 Block fuel reduction. - The alternate IDEA concepts permit changes in aircraft design philosophy, of which the most significant is the possibility of reducing the maximum limit maneuver load factor from 2.5 to 2.0. Additional design changes to the Alternate IDEA include reduction of landing sink rate, use of air cycle ECS, and improvements in EMAS and power generation systems. These produce 3.4 percent reduction in block fuel referenced to the IDEA configuration, of which 3.0 percent is attributed to the 2.0 g design load factor.

14.2.2 DOC reduction.- The reduction in DOC from the Alternate IDEA concepts is 3.1 percent. This results from the effect of downsizing the aircraft and block fuel reductions.

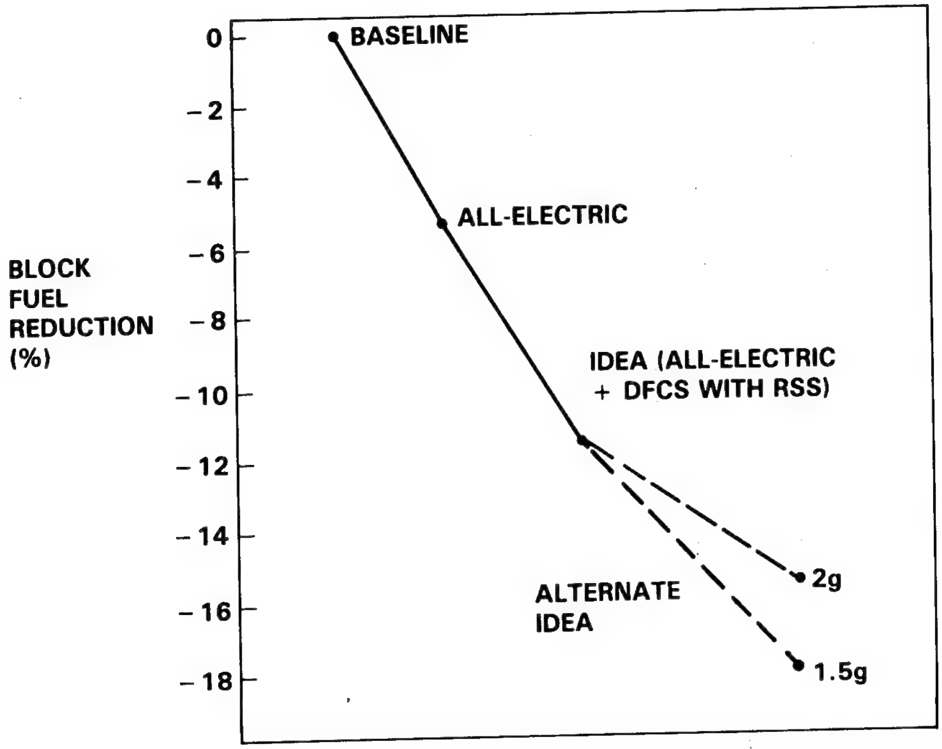


Figure 157. - Addition of technologies - impact on block fuel.

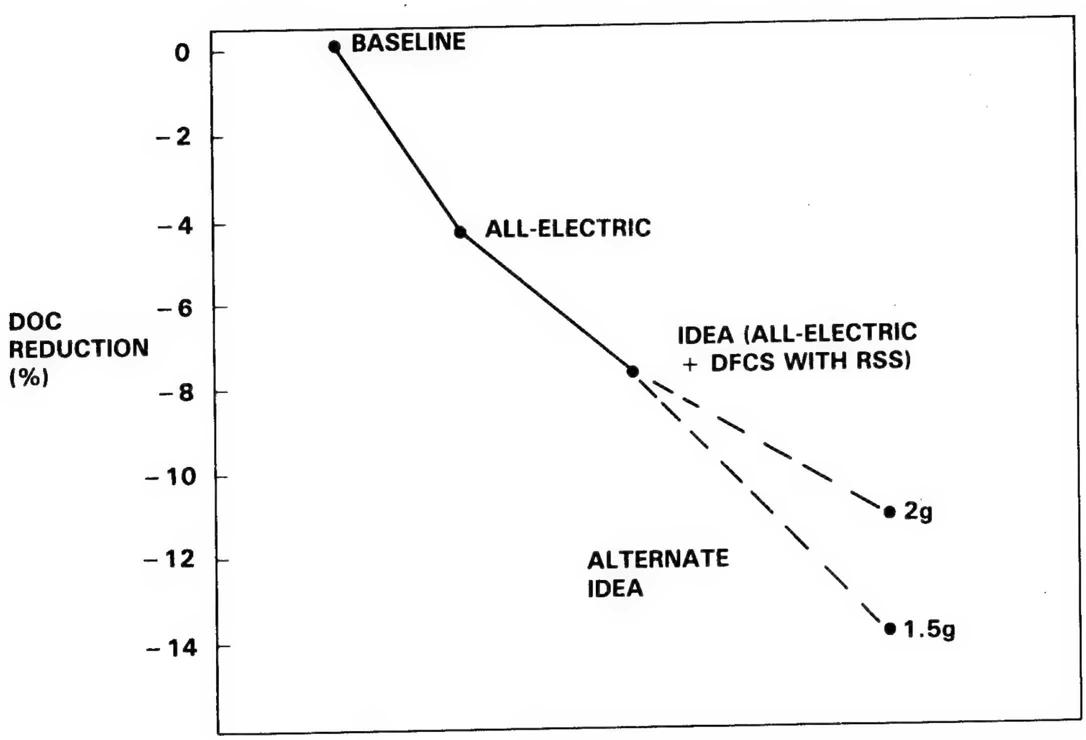


Figure 158. - Addition of technologies, impact on direct opeating cost.

14.3 Summary of Critical Technologies

Technologies that are critical to the achievement of technology readiness of the IDEA concepts are highlighted in figure 159.

14.3.1 Power generation/motor technology. - For the most part this technology is close to technology readiness, but some areas could benefit from government research, in particular:

- Improved Magnetic Material. - Samarium-cobalt permanent magnet materials offer significant benefits in electric machine weight and power density. However, cobalt based permanent magnet materials are extremely expensive and the supply of cobalt from foreign sources is subject to interruption in case of hostilities. Neodynium-iron offers the potential for much lower cost high energy product permanent magnet material (30 million gauss-oesteds), but is presently unsuitable for aerospace machines because of its operating temperature limitation. Although research into commercial applications of this material is being performed in Japan and the U.S., no concentrated effort is underway in this country to upgrade the temperature characteristics of iron based permanent magnet materials. The aerospace market alone may be too small to attract technology investment by industry.

- Higher Temperature Insulation. - As the power density of motors and generators increases, problems relative to higher operating temperature and heat dissipation become more acute. At present 20,000 hour life electrical insulation suitable for motor/generator applications are satisfactory up to 220°C (428°F) but materials rated up to 300°C (572°F) will be required.

- Higher Temperature Electronics. - A similar problem exists with electronics required for motor controllers. Currently rated for junction temperatures around 110°C (230°F), design ratings up to 150°C (302°F) will offer the option of higher temperature operating environments, and/or increased reliability.

- Innovative Cooling Mechanisms. - New methods may have to be found for cooling high power density generators/motors, such as phase change convective cooling e.g., heat pipes.

14.3.2 Engine starting. - The same problems of heat sensitivity exist with the power electronics.

14.3.3 Electric ECS. - The technology for an electrically driven air-cycle and vapor-cycle ECS is near readiness. Industry should develop these systems.

14.3.4 Power distribution. - The reliability of the power distribution system is critical to the safety of flight and demonstration of this reliability will require government participation.

TECHNOLOGY ELEMENTS \ STATUS AND REQUIREMENTS	A: NEAR TECHNOLOGY READINESS B: INDUSTRY SHOULD DEVELOP C: GOVERNMENT GUIDANCE DESIRABLE D: GOVERNMENT RESEARCH BENEFICIAL			
	A	B	C	D
POWER GENERATION/MOTOR TECHNOLOGY	X			
• IMPROVED MAGNETIC MATERIALS				X
• HIGHER TEMPERATURE INSULATION				X
• HIGHER TEMPERATURE ELECTRONICS				X
• INNOVATIVE COOLING MECHANISMS				X
ENGINE STARTING				
• POWER ELECTRONICS	X			X
ELECTRIC ECS	X	X		
POWER DISTRIBUTION				
• ELECTRIC LOAD MANAGEMENT		X	X	
• DISCRIMINATORY FAULT SENSING				X
• HIGH-SPEED FAULT CLEARING/ SYSTEM RECONFIGURATION				X
• FAULT SURVIVABILITY				X
POWER-BY-WIRE				
• ELECTRIC SYSTEM CONTROL AND PROTECTION		X		
• THERMAL MANAGEMENT				X
HIGH FREQUENCY POWER SYSTEMS				
• POWER TRANSMISSION				
– POWER LOSSES				X
– CONNECTORS				X
– RF INTERFERENCE				X
VARIABLE VOLTAGE/VARIABLE FREQUENCY				
• COMPONENT DEVELOPMENT				X
SOLID STATE ELECTRONIC LOGIC		X	X	
SOLID STATE POWER CONTROLLERS			X	
• ADEQUATE CAPACITY				X
• IMPROVED COOLING MECHANISMS				X
EMP PROTECTION (MILITARY APPLICATIONS)				X
LIGHTNING PROTECTION				X

Figure 159. - Summary of recommendations.

- Electric Load Management. - Although this technology should be developed by industry, government guidance may be desirable.

- Discriminatory Fault Sensing. - Present fault sensing techniques provide discrimination for line-to-line, line-to-neutral, and differential current, but basically rely on overcurrent or phase sequence current reactions. As a result, branch circuit faults needlessly cause the disconnection of large portions of the electric system. A technology is needed which will enable the aircraft system to be protected by disconnecting only the faulted branch circuit or individual load.

- High-Speed Fault Clearing/System Reconfiguration. - At present, sensing and clearing a fault requires 20-30 ms or greater, although it is theoretically possible to clear faults in one-half of a cycle (1.25 ms at 400 Hz).

- Fault Survivability. - At present a single fault in a paralleled generator system can cause the momentary complete loss of electric power in the aircraft and, in a number of documented cases, caused the permanent loss of electric power and subsequent loss of the aircraft.

  It is theoretically possible through a combination of system control and protection techniques to obtain the benefit of paralleled generator systems without the risk of single or multiple fault threat to flight safety.

14.3.5 High frequency power systems. - The 20 kHz system developed by NASA-Lewis is novel but untested. The major concern with this system is in power transmission, specifically.

- Power losses
- Connectors
- Radio frequency interference

14.3.6 Variable voltage/variable frequency. - The variable voltage/variable frequency power system as proposed for the IDEA configuration does not represent a high technical risk. Nevertheless, it is unlikely that a commitment could be made from component manufacturers without a clear signal from the government that the VV/VF system is the preferred choice.

14.3.7 Power-by wire.

- Electric System Control and Protection

- Thermal Management. - Heat dissipation in a composite structure raises special problems. The solution is outside the control of the actuator manufacturer and may be considered outside the area of responsibility of the composite material supplier. Possible solutions are the development of thermally conductive composites, forced air cooling, liquid loop cooling, or some other conductive means such as heat pipes on which the actuator is mounted.

14.3.8 Advanced Digital Flight Control System. - The mesh network selected for the FCS lacks sufficient maturity to develop the complex bus management software, develop specific designs, and develop design assurance methods for system validation and certification. On the other hand, the development of mesh network technology is accelerating because of ongoing work which involves prototype systems for testing.

APPENDIX A

SELECTED OUTPUT FROM AIRCRAFT PERFORMANCE AND SIZING PROGRAM (ASSET)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

C O N F I G U R A T I O N   G E O M E T R Y

BASIC WING--	AREA(SQ FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	MAC(FT)		
	3589.5	207.54	0.246	25.000	27.327	19.41		
WING PANELS--	AREA(SQ FT)	EXP. AREA	AVG T/C	L.E. SWEEP	SFLE(SQ FT)	REF L(FT)		
	782.5	62.1	9.29	27.327	0.0	34.83		
	1818.6	1818.6	10.00	27.327	0.0	27.48		
	261.9	261.9	11.16	27.327	0.0	18.04		
	1142.3	1142.3	10.45	27.327	0.0	13.33		
	110.0	110.0	9.16	27.327	0.0	7.59		
TOTAL WING--	AREA(SQ FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	MAC(FT)	L(FT)	E SWP
	4115.3	10.47	10.02	39.53	6.84	24.25	60.46	16.389
FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)			
	201.00	10603.5	19.32	19.58	301.21			
	BW(FT)	BH(FT)	SBW(SQ FT)					
	19.58	19.58	10538.00					
HORZ. TAIL 1--	SHT1(SQ FT)	SHX1(SQ FT)	REF L1(FT)	L HT1(FT)	HT1 VOL COEF			
	955.00	719.85	13.57	78.76	1.0797			
HORZ. TAIL 2--	SHT2(SQ FT)	SHX2(SQ FT)	REF L2(FT)	L HT2(FT)	HT2 VOL COEF			
	0.0	0.0	0.0	201.00	0.0			
VERT. TAIL 1--	SVT1(SQ FT)	SVX1(SQ FT)	REF L1(FT)	L VT1(FT)	VT1 VOL COEF			
	567.46	334.28	17.21	76.91	0.0586			
VERT. TAIL 2--	SVT2(SQ FT)	SVX2(SQ FT)	REF L2(FT)	L VT2(FT)	VT2 VOL COEF			
	0.0	0.0	0.0	201.00	0.0			
PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)	
	15.41	6.90	19.64	8.13	937.28	2.	20.21	
FUEL TANKS--	WING(CU FT)	BOX(CU FT)	FUS(CU FT)					
	2612.07	496.44	999999.00					
WETTED VOLUMES--	BODY	WING	TAILS	PODS	PYLONS	PONTOONS	TOTAL	
	50311.41	4853.41	1343.52	1726.05	146.65	0.0	58381.03	

Baseline

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(470224.)		
FUEL AVAILABLE	142908.	FUEL	30.39
EXTERNAL	0.		
INTERNAL	142916.		
ZERO FUEL WEIGHT	327316.		
PAYLOAD	73500.	PAYLOAD	15.63
PASSENGERS	57750.		
BAGGAGE	15750.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	253816.		
OPERATIONAL ITEMS	10319.	OPERATIONAL ITEMS	4.23
STANDARD ITEMS	9557.		
EMPTY WEIGHT	233940.		
STRUCTURE	143410.	STRUCTURE	30.50
WING	53785.		
ROTOR	0.		
TAIL	5579.		
BODY	55108.		
ALIGHTING GEAR	19601.		
ENGINE SECTION AND NACELLE	9337.		
PROPULSION	26004.	PROPULSION	5.53
CRUISE ENGINES	20422.		
LIFT ENGINES	0.		
THRUST REVERSER	3176.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	220.		
STARTING SYSTEM	536.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	1650.		
DRIVE SYSTEM (POWER TRANS)	0.		
SYSTEMS	64527.	SYSTEMS	13.72
FLIGHT CONTROLS	5803.		
AUXILIARY POWER PLANT	1202.		
INSTRUMENTS	1058.		
HYDRAULIC AND PNEUMATIC	2823.		
ELECTRICAL	5749.		
AVIONICS	2827.		
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	38504.		
AIR CONDITIONING	6182.		
ANTI-ICING	379.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

Baseline

M I S S I O N   S U M M A R Y

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	470224.	0.	0.	0.	0.	0.0	0.0	0.	745701.	0.	0.0	0.0	0.615
POWER 2	0.	0.0	470224.	572.	572.	0.	0.	1.0	1.0	0.	745401.	0.	0.0	0.0	0.304
CLIMB	0.	0.378	469652.	2528.	3100.	27.	27.	5.9	6.9	0.	745201.	0.	0.617	19.83	0.462
ACCEL	10000.	0.456	467124.	387.	3486.	5.	32.	1.0	7.9	0.	745201.	0.	0.509	19.21	0.489
CLIMB	10000.	0.547	466738.	13128.	16614.	333.	365.	45.1	53.0	0.	745201.	0.	0.455	18.36	0.546
CRUISE	35000.	0.800	453610.	0.	16614.	0.	365.	0.0	53.0	0.	-745101.	0.	0.565	19.43	0.546
ACCEL	35000.	0.800	453610.	0.	16614.	0.	365.	0.0	53.0	0.	745201.	0.	0.565	19.43	0.547
ACCEL	35000.	0.800	453610.	0.	16614.	0.	365.	0.0	53.0	0.	745201.	0.	0.565	19.43	0.547
CRUISE	35000.	0.800	453610.	99918.	116532.	4035.	4400.	527.4	580.4	0.	-745101.	0.	0.566	19.27	0.545
CLIMB	40000.	0.800	353691.	0.	116532.	0.	4400.	0.0	580.4	0.	745201.	0.	0.0	0.0	0.0
CRUISE	40000.	0.800	353691.	1.	116533.	0.	4400.	0.0	580.4	0.	-745101.	0.	0.559	19.08	0.547
CLIMB	40000.	0.800	353691.	0.	116533.	0.	4400.	0.0	580.4	0.	745201.	0.	0.0	0.0	0.0
CRUISE	40000.	0.800	353691.	0.	116533.	0.	4400.	0.0	580.4	0.	-745101.	0.	0.559	19.08	0.547
DESCENT	40000.	0.800	353690.	1213.	117747.	117.	4517.	16.1	596.5	0.	745503.	0.	0.390	16.60	-0.278
DECEL	10000.	0.547	352477.	43.	117790.	4.	4521.	0.8	597.4	0.	745503.	0.	0.389	17.07	-5.274
DESCENT	10000.	0.456	352434.	387.	118177.	29.	4551.	6.5	603.9	0.	745503.	0.	0.464	18.56	-1.218
CRUISE	40000.	0.800	352046.	1087.	119264.	49.	4600.	6.5	610.3	0.	-745101.	0.	0.556	19.05	0.547
LOITER	1500.	0.295	350959.	445.	119708.	0.	4600.	3.0	613.3	0.	-745101.	0.	0.800	19.49	0.494
CRUISE	1500.	0.378	350515.	342.	120050.	0.	4600.	2.0	615.3	0.	-745101.	0.	0.487	18.89	0.552
RESET	0.	0.0	350173.	0.	120050.	-4600.	0.	0.0	615.3	0.	0.	0.	0.0	0.0	0.0
CRUISE	40000.	0.800	350173.	10227.	130277.	0.	0.	61.5	676.9	0.	-745101.	0.	0.546	18.96	0.548
TAKEOFF POWER 1	0.	0.0	339946.	0.	130277.	0.	0.	0.0	676.9	0.	745701.	0.	0.0	0.0	0.615
POWER 2	0.	0.0	339946.	572.	130849.	0.	0.	1.0	677.9	0.	745401.	0.	0.0	0.0	0.304
CLIMB	0.	0.378	339374.	1575.	132424.	17.	17.	3.7	681.6	0.	745201.	0.	0.445	18.22	0.462
ACCEL	10000.	0.456	337799.	242.	132666.	3.	20.	0.6	682.2	0.	745201.	0.	0.368	16.53	0.489

Baseline

CLIMB	10000.	0.547	337557.	3895.	136560.	80.	100.	11.7	693.9	0.	745201.	0.	0.309	14.91	0.534
CRUISE	30000.	0.700	333663.	0.	136560.	0.	100.	0.0	693.9	0.	-745101.	0.	0.430	17.79	0.540
DESCENT	30000.	0.750	333663.	308.	136868.	51.	151.	7.6	701.6	0.	745503.	0.	0.311	14.95	-1.025
DECEL	10000.	0.547	333355.	42.	136910.	4.	156.	0.8	702.4	0.	745503.	0.	0.368	16.53	-5.288
DESCENT	10000.	0.456	333313.	378.	137288.	29.	184.	6.4	708.7	0.	745503.	0.	0.439	18.10	-1.211
CRUISE	30000.	0.695	332935.	377.	137665.	15.	200.	2.3	711.0	0.	-745101.	0.	0.435	17.88	0.539
CRUISE	1500.	0.375	332558.	5286.	142951.	0.	200.	32.0	743.0	0.	-745101.	0.	0.466	18.55	0.557

WTO = 470224.0 FUEL A=142907.8 FUEL R=142951.1

Baseline

ALTERNATE MISSION NO. 1 SUMMARY

ATX-350I, AVRG STAGE L = 2500 NMI, M = 0.80, 65% LF

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	378476.	0.	0.	0.	0.	0.0	0.0	0.	745701.	0.	0.0	0.0	0.615
POWER 2	0.	0.0	378476.	572.	572.	0.	0.	1.0	1.0	0.	745401.	0.	0.0	0.0	0.304
CLIMB	0.	0.378	377904.	1815.	2387.	19.	19.	4.2	5.2	0.	745201.	0.	0.496	19.04	0.462
ACCEL	10000.	0.456	376089.	279.	2666.	4.	23.	0.7	6.0	0.	745201.	0.	0.410	17.56	0.489
CLIMB	10000.	0.547	375810.	9363.	12029.	251.	273.	34.0	39.9	0.	745201.	0.	0.401	16.92	0.545
CRUISE	39000.	0.800	366447.	0.	12029.	0.	273.	0.0	39.9	0.	-745101.	0.	0.553	19.09	0.547
ACCEL	39000.	0.800	366447.	0.	12029.	0.	273.	0.0	39.9	0.	745201.	0.	0.553	19.09	0.547
ACCEL	39000.	0.800	366447.	0.	12029.	0.	273.	0.0	39.9	0.	745201.	0.	0.553	19.09	0.547
CRUISE	39000.	0.800	366447.	43929.	55957.	2027.	2300.	265.0	304.9	0.	-745101.	0.	0.547	18.96	0.548
CLIMB	41000.	0.800	322518.	0.	55957.	0.	2300.	0.0	304.9	0.	745201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	322518.	0.	55957.	0.	2300.	0.0	304.9	0.	-745101.	0.	0.535	18.79	0.550
CLIMB	41000.	0.800	322518.	0.	55957.	0.	2300.	0.0	304.9	0.	745201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	322518.	0.	55957.	0.	2300.	0.0	304.9	0.	-745101.	0.	0.535	18.79	0.550
DESCENT	41000.	0.800	322518.	1417.	57375.	126.	2426.	17.2	322.1	0.	745503.	0.	0.365	15.88	-0.146
DECEL	10000.	0.547	321101.	41.	57416.	4.	2430.	0.8	322.9	0.	745503.	0.	0.355	16.16	-5.297
DESCENT	10000.	0.456	321060.	371.	57787.	28.	2458.	6.2	329.2	0.	745503.	0.	0.423	17.76	-1.206
CRUISE	41000.	0.800	320688.	853.	58640.	42.	2500.	5.4	334.6	0.	-745101.	0.	0.531	18.76	0.550
LOITER	1500.	0.280	319835.	407.	59047.	0.	2500.	3.0	337.6	0.	-745101.	0.	0.810	19.40	0.495
CRUISE	1500.	0.378	319428.	329.	59377.	0.	2500.	2.0	339.6	0.	-745101.	0.	0.444	18.17	0.562
RESET	0.	0.0	319099.	0.	59377.	-2500.	0.	0.0	339.6	0.	0.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	319099.	5278.	64654.	0.	0.	34.0	373.6	0.	-745101.	0.	0.525	18.68	0.551
TAKEOFF POWER 1	0.	0.0	313821.	0.	64654.	0.	0.	0.0	373.6	0.	745701.	0.	0.0	0.0	0.615
POWER 2	0.	0.0	313821.	572.	65226.	0.	0.	1.0	374.6	0.	745401.	0.	0.0	0.0	0.304
CLIMB	0.	0.378	313249.	1426.	66652.	15.	15.	3.3	377.9	0.	745201.	0.	0.409	17.46	0.462
ACCEL	10000.	0.456	311823.	218.	66870.	3.	18.	0.6	378.5	0.	745201.	0.	0.340	15.72	0.489

Baseline

CLIMB	10000.	0.547	311605.	3866.	70736.	82.	100.	11.9	390.4	0.	745201.	0.	0.290	14.22	0.535
CRUISE	30000.	0.665	307739.	0.	70736.	0.	100.	0.0	390.4	0.	-745101.	0.	0.440	17.87	0.533
DESCENT	30000.	0.750	307739.	290.	71026.	48.	148.	7.2	397.6	0.	745503.	0.	0.287	14.10	-1.025
DECEL	10000.	0.547	307449.	40.	71065.	4.	152.	0.8	398.3	0.	745503.	0.	0.340	15.72	-5.307
DESCENT	10000.	0.456	307410.	363.	71428.	28.	180.	6.1	404.4	0.	745503.	0.	0.405	17.35	-1.201
CRUISE	30000.	0.665	307047.	463.	71892.	20.	200.	3.0	407.5	0.	-745101.	0.	0.439	17.84	0.533
CRUISE	1500.	0.375	306583.	5128.	77020.	0.	200.	32.0	439.5	0.	-745101.	0.	0.429	17.86	0.565

Baseline

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.50906	13.63673
FUEL AND OIL	1.67290	44.81335
INSURANCE	0.04484	1.20128
DEPRECIATION	0.80809	21.64685
MAINTENANCE	0.69815	18.70177
TOTAL DOC	3.73304	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.32030	11.53374
AIRCRAFT CONTROL	0.00854	0.30762
CABIN ATTENDANT	0.50650	18.23843
FOOD AND BEVERAGE	0.18142	6.53286
PASSENGER HANDLING	0.41303	14.87282
CARGO HANDLING	0.14903	5.36638
OTHER PASSENGER EXPENSE	0.99450	35.81073
OTHER CARGO EXPENSE	0.02268	0.81676
GENERAL + ADMINISTRATION	0.18109	6.52075
TOTAL IOC	2.77710	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	4600.00
BLOCK FUEL (LBS)	120801.00
BLOCK TIME (HRS)	10.59
FLIGHT TIME (HRS)	10.26
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4618.50
FLIGHTS PER A/C PER YEAR	436.16
FARE ($)	462.19
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	525.40	29.55	495.85	241.44	63.05	228.57	-256.01	2.59
2	13.0	8.0	1366.05	106.39	1259.65	627.74	153.84	594.29	-346.18	2.66
3	20.4	7.0	2141.02	226.83	1914.19	983.86	226.66	931.44	-394.27	2.74
4	23.0	0.0	2416.85	362.77	2054.08	1110.62	230.76	1051.44	-228.98	2.83
5	23.0	0.0	2416.85	498.72	1918.13	1110.62	201.75	1051.44	-199.98	3.09
6	23.0	0.0	2416.85	634.67	1782.18	1110.62	172.75	1051.44	-170.98	3.32
7	23.0	0.0	2416.85	770.62	1646.24	1110.62	143.75	1051.44	-141.98	3.59
8	23.0	0.0	2416.85	906.56	1510.29	1110.62	114.75	1051.44	-112.97	3.92
9	23.0	0.0	2416.85	1042.51	1374.34	1110.62	85.75	1051.44	-83.97	4.31
10	23.0	0.0	2416.85	1178.46	1238.39	1110.62	56.74	1051.44	-56.18	4.68
11	23.0	0.0	2416.85	1314.41	1102.45	1110.62	27.74	1051.44	25.57	3.94
12	23.0	0.0	2416.85	1450.35	966.50	1110.62	8.83	1051.44	102.28	3.52
13	23.0	0.0	2416.85	1586.30	830.55	1110.62	-0.00	1051.44	165.54	3.56
14	23.0	0.0	2416.85	1722.25	694.60	1110.62	-0.00	1051.44	165.54	4.26
15	23.0	0.0	2416.85	1858.20	558.66	1110.62	-0.00	1051.44	165.54	5.30
16	23.0	0.0	2416.85	1994.14	422.71	1110.62	-0.00	1051.44	165.54	7.00

AVG ROI OVER THE 16 YEAR PERIOD = 3.58 PERCENT

Baseline

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.53016	13.80390
FUEL AND OIL	1.53218	39.89351
INSURANCE	0.05207	1.35565
DEPRECIATION	0.93822	24.42856
MAINTENANCE	0.78804	20.51840
TOTAL DOC	3.84067	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.58936	16.46843
AIRCRAFT CONTROL	0.01572	0.43923
CABIN ATTENDANT	0.52749	14.73968
FOOD AND BEVERAGE	0.18894	5.27963
PASSENGER HANDLING	0.75998	21.23611
CARGO HANDLING	0.27421	7.66238
OTHER PASSENGER EXPENSE	0.99450	27.78929
OTHER CARGO EXPENSE	0.02268	0.63381
GENERAL + ADMINISTRATION	0.20583	5.75153
TOTAL IOC	3.57871	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	2500.00
BLOCK FUEL (LBS)	60127.50
BLOCK TIME (HRS)	5.99
FLIGHT TIME (HRS)	5.66
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4142.75
FLIGHTS PER A/C PER YEAR	691.22
FARE ($)	299.45
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

- -

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	525.40	29.55	495.85	245.89	63.05	224.37	-247.35	4.34
2	13.0	8.0	1366.05	106.39	1259.65	639.32	153.84	583.35	-323.66	4.44
3	20.4	7.0	2141.02	226.83	1914.19	1002.02	226.66	914.29	-358.96	4.58
4	23.0	0.0	2416.85	362.77	2054.08	1131.11	230.76	1032.09	-189.13	4.82
5	23.0	0.0	2416.85	498.72	1918.13	1131.11	201.75	1032.09	-160.13	5.16
6	23.0	0.0	2416.85	634.67	1782.18	1131.11	172.75	1032.09	-131.13	5.56
7	23.0	0.0	2416.85	770.62	1646.24	1131.11	143.75	1032.09	-102.13	6.02
8	23.0	0.0	2416.85	906.56	1510.29	1131.11	114.75	1032.09	-73.12	6.56
9	23.0	0.0	2416.85	1042.51	1374.34	1131.11	85.75	1032.09	-50.76	6.72
10	23.0	0.0	2416.85	1178.46	1238.39	1131.11	56.74	1032.09	-36.26	6.29
11	23.0	0.0	2416.85	1314.41	1102.45	1131.11	27.74	1032.09	45.49	5.75
12	23.0	0.0	2416.85	1450.35	966.50	1131.11	8.83	1032.09	122.20	5.58
13	23.0	0.0	2416.85	1586.30	830.55	1131.11	-0.00	1032.09	185.46	5.96
14	23.0	0.0	2416.85	1722.25	694.60	1131.11	-0.00	1032.09	185.46	7.13
15	23.0	0.0	2416.85	1858.20	558.66	1131.11	-0.00	1032.09	185.46	8.86
16	23.0	0.0	2416.85	1994.14	422.71	1131.11	-0.00	1032.09	185.46	11.71

AVG ROI OVER THE 16 YEAR PERIOD = 5.80 PERCENT

Baseline

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S MAY 24 1984

AIRCRAFT MODEL --CL1633-11-1
I.O.C. DATE --1990
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 745000
SLS SCALE 1.0 = 39000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 25.00 DEG
WING TAPER RATIO = 0.246

1 W/S	131.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.249	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	25.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4600	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	470224	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	142908	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	253816	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	327316	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	1.001	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	39029	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3539.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	207.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	955.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	567.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	15.41	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	6.90	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	201.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.088	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	3.284	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	89.97	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	31.524	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	3.733	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.777	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - O/O	3.58	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	35000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	120801	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
34 MISN V1(2,1)	39000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 MISN V2(2,1)	60128	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
36 TAKEOFF DST(1)	9291	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(1)	0.1074	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 TAKEOFF DST(2)	9410	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 CLIMB GRAD(2)	0.0432	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
40 CTOL LNDG D(1)	6076	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
41 AP SPEED-KT(1)	134.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
42 CTOL LNDG D(2)	6683	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
43 AP SPEED-KT(2)	143.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
44 CTOL LNDG D(3)	7289	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
45 AP SPEED-KT(3)	152.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
46 SEP(1) - FPS	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
47 SEP(2) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
48 SEP(3) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
49 SEP(4) - FPS	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Baseline

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ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00       AR=12.00        W/S=131.00       T/W=0.249

            C O N F I G U R A T I O N   G E O M E T R Y

BASIC WING--      AREA(SQ FT)   SPAN(FT)   TAPER RATIO  C/4 SWEEP  L.E. SWEEP    MAC(FT)
                     3589.5      207.54       0.246       25.000     27.327       19.41

WING PANELS--     AREA(SQ FT)   EXP. AREA    AVG T/C    L.E. SWEEP SFLE(SQ FT)  REF L(FT)
                      782.5        62.1        9.29       27.327      0.0          34.83
                     1818.6      1818.6       10.00       27.327      0.0          27.48
                      261.9       261.9       11.16       27.327      0.0          18.04
                     1142.3      1142.3       10.45       27.327      0.0          13.33
                      110.0       110.0        9.16       27.327      0.0           7.59

TOTAL WING--      AREA(SQ FT)    EFF AR      AVG T/C     CR(FT)      CT(FT)      MAC(FT)     L(FT)     E SWP
                     4115.3       10.47       10.02       39.53        6.84       24.25       60.46    16.389

FUSELAGE--        LENGTH(FT)  S WET(SQ FT)   BWH(FT)   EQUIV D(FT) SPI(SQ FT)
                     201.00     10603.5       19.32       19.58      301.21

                     BW(FT)      BH(FT)     SBW(SQ FT)
                      19.58       19.58     10538.00

HORZ. TAIL 1--    SHT1(SQ FT) SHX1(SQ FT)  REF L1(FT)   L HT1(FT)  HT1 VOL COEF
                     548.40      373.81        9.90       78.81      0.6204

HORZ. TAIL 2--    SHT2(SQ FT) SHX2(SQ FT)  REF L2(FT)   L HT2(FT)  HT2 VOL COEF
                        0.0         0.0         0.0      201.00      0.0

VERT. TAIL 1--    SVT1(SQ FT) SVX1(SQ FT)  REF L1(FT)   L VT1(FT)  VT1 VOL COEF
                     567.46      334.28       17.21       76.91      0.0586

VERT. TAIL 2--    SVT2(SQ FT) SVX2(SQ FT)  REF L2(FT)   L VT2(FT)  VT2 VOL COEF
                        0.0         0.0         0.0      201.00      0.0

PROPULSION--        ENG L(FT)   ENG D(FT)   POD L(FT)   POD D(FT)   POD S WET    NO. PODS  INLET L(FT)
                      15.41        6.90       19.64        8.13      937.28         2.        20.21

FUEL TANKS--      WING(CU FT)  BOX(CU FT)  FUS(CU FT)
                    2612.07      496.44    999999.00

WETTED VOLUMES-- BODY           WING        TAILS        PODS       PYLONS     PONTOONS     TOTAL
                   50311.41     4853.41      777.55     1726.05      146.65       0.0      57815.07
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IDEA (Unsized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

WEIGHT STATEMENT

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(470224.)		
FUEL AVAILABLE	152621.	FUEL	32.46
EXTERNAL	0.		
INTERNAL	152623.		
ZERO FUEL WEIGHT	317603.		
PAYLOAD	73500.	PAYLOAD	15.63
PASSENGERS	57750.		
BAGGAGE	15750.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	244103.		
OPERATIONAL ITEMS	10319.	OPERATIONAL ITEMS	4.23
STANDARD ITEMS	9569.		
EMPTY WEIGHT	224214.		
STRUCTURE	140703.	STRUCTURE	29.92
WING	53791.		
ROTOR	0.		
TAIL	3805.		
BODY	54688.		
ALIGHTING GEAR	19186.		
ENGINE SECTION AND NACELLE	9233.		
PROPULSION	26019.	PROPULSION	5.53
CRUISE ENGINES	20129.		
LIFT ENGINES	0.		
THRUST REVERSER	2831.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	100.		
STARTING SYSTEM	140.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2818.		
DRIVE SYSTEM (POWER TRANS)	0.		
SYSTEMS	57492.		
FLIGHT CONTROLS	4399.		
AUXILIARY POWER PLANT	962.		
INSTRUMENTS	1058.		
HYDRAULIC AND PNEUMATIC	0.		
ELECTRICAL	5446.		
AVIONICS	2827.	SYSTEMS	12.23
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	38504.		
AIR CONDITIONING	3848.		
ANTI-ICING	448.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

IDEA (Unsized)

MISSION SUMMARY

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	470224.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	470224.	576.	576.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	469648.	2408.	2984.	25.	25.	5.6	6.6	0.	751201.	0.	0.617	20.59	0.458
ACCEL	10000.	0.456	467240.	362.	3347.	5.	30.	0.9	7.5	0.	751201.	0.	0.510	19.95	0.484
CLIMB	10000.	0.547	466877.	11706.	15052.	298.	328.	40.5	48.0	0.	751201.	0.	0.476	19.22	0.538
CRUISE	37000.	0.800	455172.	0.	15052.	0.	328.	0.0	48.0	0.	-751101.	0.	0.624	20.30	0.538
ACCEL	37000.	0.800	455172.	0.	15052.	0.	328.	0.0	48.0	0.	751201.	0.	0.624	20.30	0.538
ACCEL	37000.	0.800	455172.	0.	15052.	0.	328.	0.0	48.0	0.	751201.	0.	0.624	20.30	0.538
CRUISE	37000.	0.800	455171.	96257.	111309.	4072.	4400.	532.3	580.4	0.	-751101.	0.	0.607	20.15	0.539
CLIMB	41000.	0.800	358915.	0.	111309.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	358915.	0.	111309.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.596	19.98	0.542
CLIMB	41000.	0.800	358914.	0.	111309.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	358914.	0.	111309.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.596	19.98	0.542
DESCENT	41000.	0.800	358914.	1086.	112396.	119.	4519.	16.5	596.9	0.	751503.	0.	0.402	17.42	-0.338
DECEL	10000.	0.547	357828.	43.	112439.	5.	4524.	0.9	597.8	0.	751503.	0.	0.395	17.86	-18.175
DESCENT	10000.	0.456	357785.	390.	112829.	31.	4555.	6.9	604.6	0.	751503.	0.	0.471	19.38	6.545
CRUISE	41000.	0.800	357395.	946.	113775.	45.	4600.	5.9	610.5	0.	-751101.	0.	0.592	19.96	0.542
LOITER	1500.	0.295	356449.	436.	114211.	0.	4600.	3.0	613.5	0.	-751101.	0.	0.813	20.10	0.492
CRUISE	1500.	0.378	356013.	333.	114543.	0.	4600.	2.0	615.5	0.	-751101.	0.	0.495	19.71	0.553
RESET	0.	0.0	355680.	0.	114543.	-4600.	0.	0.0	615.5	0.	0.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	355680.	9807.	124350.	0.	0.	61.5	677.0	0.	-751101.	0.	0.582	19.90	0.542
TAKEOFF POWER 1	0.	0.0	345873.	0.	124350.	0.	0.	0.0	677.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	345873.	576.	124926.	0.	0.	1.0	678.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	345297.	1553.	126480.	16.	16.	3.6	681.6	0.	751201.	0.	0.452	19.04	0.458
ACCEL	10000.	0.456	343744.	235.	126715.	3.	19.	0.6	682.2	0.	751201.	0.	0.375	17.33	0.484

IDEA (Unsized)

CLIMB	10000.	0.547	343508.	3900.	130614.	81.	100.	11.8	694.0	0.	751201.	0.	0.318	15.75	0.529
CRUISE	30000.	0.700	339609.	0.	130614.	0.	100.	0.0	694.0	0.	-751101.	0.	0.438	18.62	0.538
DESCENT	30000.	0.750	339609.	310.	130924.	54.	154.	8.1	702.0	0.	751503.	0.	0.316	15.69	-1.053
DECEL	10000.	0.547	339299.	42.	130966.	5.	159.	0.8	702.9	0.	751503.	0.	0.375	17.33	-18.266
DESCENT	10000.	0.456	339257.	382.	131348.	30.	189.	6.7	709.6	0.	751503.	0.	0.446	18.93	6.542
CRUISE	30000.	0.695	338875.	256.	131603.	11.	200.	1.6	711.2	0.	-751101.	0.	0.443	18.71	0.536
CRUISE	1500.	0.378	338619.	5197.	136801.	0.	200.	32.0	743.2	0.	-751101.	0.	0.467	19.28	0.559

WTO = 470224.0 FUEL A=152621.4 FUEL R=136800.6

IDEA (Unsized)

A L T E R N A T E   M I S S I O N   N O. 1   S U M M A R Y

ATX-350I, AVRG STAGE L = 2500 NMI, M = 0.80, 65% LF

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	363134.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	363134.	576.	576.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	362558.	1653.	2230.	17.	17.	3.8	4.8	0.	751201.	0.	0.475	19.46	0.458
ACCEL	10000.	0.456	360905.	250.	2480.	3.	21.	0.6	5.5	0.	751201.	0.	0.393	17.82	0.484
CLIMB	10000.	0.547	360654.	8074.	10554.	220.	240.	29.8	35.3	0.	751201.	0.	0.408	17.46	0.538
CRUISE	41000.	0.800	352580.	0.	10554.	0.	240.	0.0	35.3	0.	-751101.	0.	0.585	19.92	0.542
ACCEL	41000.	0.800	352580.	0.	10554.	0.	240.	0.0	35.3	0.	751201.	0.	0.585	19.92	0.541
ACCEL	41000.	0.800	352580.	0.	10554.	0.	240.	0.0	35.3	0.	751201.	0.	0.585	19.92	0.541
CRUISE	41000.	0.800	352580.	40923.	51477.	2060.	2300.	269.3	304.5	0.	-751101.	0.	0.578	19.80	0.544
CLIMB	43000.	0.800	311657.	0.	51477.	0.	2300.	0.0	304.5	0.	751201.	0.	0.0	0.0	0.0
CRUISE	43000.	0.800	311657.	0.	51477.	0.	2300.	0.0	304.5	0.	-751101.	0.	0.569	19.68	0.547
CLIMB	43000.	0.800	311657.	0.	51477.	0.	2300.	0.0	304.5	0.	751201.	0.	0.0	0.0	0.0
CRUISE	43000.	0.800	311657.	0.	51477.	0.	2300.	0.0	304.5	0.	-751101.	0.	0.569	19.68	0.547
DESCENT	43000.	0.800	311657.	1402.	52880.	136.	2436.	18.5	323.1	0.	751503.	0.	0.368	16.37	-0.116
DECEL	10000.	0.547	310254.	40.	52919.	4.	2440.	0.8	323.9	0.	751503.	0.	0.343	16.40	-18.416
DESCENT	10000.	0.456	310215.	365.	53285.	29.	2469.	6.4	330.3	0.	751503.	0.	0.408	18.09	6.537
CRUISE	43000.	0.800	309849.	580.	53864.	31.	2500.	4.0	334.4	0.	-751101.	0.	0.565	19.65	0.547
LOITER	1500.	0.275	309270.	384.	54248.	0.	2500.	3.0	337.4	0.	-751101.	0.	0.812	20.02	0.497
CRUISE	1500.	0.378	308886.	315.	54563.	0.	2500.	2.0	339.4	0.	-751101.	0.	0.429	18.53	0.567
RESET	0.	0.0	308571.	0.	54563.	-2500.	0.	0.0	339.4	0.	0.	0.	0.0	0.0	0.0
CRUISE	42001.	0.800	308571.	4876.	59440.	0.	0.	33.9	373.3	0.	-751101.	0.	0.533	19.41	0.547
TAKEOFF POWER 1	0.	0.0	303694.	0.	59440.	0.	0.	0.0	373.3	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	303694.	576.	60016.	0.	0.	1.0	374.3	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	303118.	1324.	61340.	14.	14.	3.1	377.4	0.	751201.	0.	0.395	17.78	0.458
ACCEL	10000.	0.456	301794.	200.	61540.	3.	16.	0.5	377.9	0.	751201.	0.	0.329	15.96	0.484

IDEA (Unsized)

CLIMB	10000.	0.547	301594.	3384.	64924.	71.	87.	10.3	388.1	0.	751201.	0.	0.281	14.37	0.529
CRUISE	30000.	0.650	298210.	284.	65208.	13.	100.	2.0	390.1	0.	-751101.	0.	0.446	18.60	0.527
DESCENT	30000.	0.750	297926.	280.	65488.	49.	149.	7.3	397.4	0.	751503.	0.	0.278	14.24	-1.052
DECEL	10000.	0.547	297646.	39.	65527.	4.	153.	0.8	398.2	0.	751503.	0.	0.329	15.96	-18.488
DESCENT	10000.	0.456	297607.	358.	65884.	29.	182.	6.3	404.5	0.	751503.	0.	0.392	17.69	6.534
CRUISE	30000.	0.650	297249.	402.	66287.	18.	200.	2.9	407.4	0.	-751101.	0.	0.444	18.57	0.527
CRUISE	1500.	0.375	296847.	4920.	71207.	0.	200.	32.0	439.4	0.	-751101.	0.	0.416	18.21	0.571

IDEA (Unsized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.50919	14.12646
FUEL AND OIL	1.59595	44.27647
INSURANCE	0.05030	1.39540
DEPRECIATION	0.79357	22.01595
MAINTENANCE	0.65551	18.18570
TOTAL DOC	3.60451	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.31992	11.53627
AIRCRAFT CONTROL	0.00854	0.30806
CABIN ATTENDANT	0.50662	18.26900
FOOD AND BEVERAGE	0.18147	6.54380
PASSENGER HANDLING	0.41303	14.89409
CARGO HANDLING	0.14903	5.37406
OTHER PASSENGER EXPENSE	0.99450	35.86195
OTHER CARGO EXPENSE	0.02268	0.81793
GENERAL + ADMINISTRATION	0.17734	6.39496
TOTAL IOC	2.77313	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	4600.00
BLOCK FUEL (LBS)	115242.50
BLOCK TIME (HRS)	10.59
FLIGHT TIME (HRS)	10.26
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4618.72
FLIGHTS PER A/C PER YEAR	436.07
FARE ($)	462.19
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

- -

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	515.81	29.01	486.80	241.39	61.90	223.88	-246.45	3.60
2	13.0	8.0	1341.11	104.45	1236.66	627.62	151.03	582.08	-327.17	3.68
3	20.4	7.0	2101.93	222.69	1879.25	983.67	222.52	912.31	-367.17	3.80
4	23.0	0.0	2372.74	356.15	2016.58	1110.40	226.54	1029.84	-202.34	3.99
5	23.0	0.0	2372.74	489.62	1883.12	1110.40	198.07	1029.84	-173.87	4.28
6	23.0	0.0	2372.74	623.08	1749.65	1110.40	169.60	1029.84	-145.39	4.60
7	23.0	0.0	2372.74	756.55	1616.19	1110.40	141.13	1029.84	-116.92	4.98
8	23.0	0.0	2372.74	890.02	1482.72	1110.40	112.65	1029.84	-88.45	5.43
9	23.0	0.0	2372.74	1023.48	1349.25	1110.40	84.18	1029.84	-59.97	5.97
10	23.0	0.0	2372.74	1156.95	1215.79	1110.40	55.71	1029.84	-43.93	5.60
11	23.0	0.0	2372.74	1290.41	1082.32	1110.40	27.23	1029.84	36.33	4.98
12	23.0	0.0	2372.74	1423.88	948.86	1110.40	8.67	1029.84	111.64	4.70
13	23.0	0.0	2372.74	1557.35	815.39	1110.40	-0.00	1029.84	173.75	4.94
14	23.0	0.0	2372.74	1690.81	681.92	1110.40	-0.00	1029.84	173.75	5.91
15	23.0	0.0	2372.74	1824.28	548.46	1110.40	-0.00	1029.84	173.75	7.34
16	23.0	0.0	2372.74	1957.75	414.99	1110.40	-0.00	1029.84	173.75	9.71

AVG ROI OVER THE 16 YEAR PERIOD = 4.87 PERCENT

IDEA (Unsized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.52980	14.49605
FUEL AND OIL	1.40825	38.53134
INSURANCE	0.05835	1.59663
DEPRECIATION	0.92068	25.19083
MAINTENANCE	0.73773	20.18524
TOTAL DOC	3.65482	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.58865	16.47963
AIRCRAFT CONTROL	0.01572	0.44006
CABIN ATTENDANT	0.52713	14.75758
FOOD AND BEVERAGE	0.18882	5.28604
PASSENGER HANDLING	0.75998	21.27629
CARGO HANDLING	0.27421	7.67687
OTHER PASSENGER EXPENSE	0.99450	27.84187
OTHER CARGO EXPENSE	0.02268	0.63501
GENERAL + ADMINISTRATION	0.20027	5.60675
TOTAL IOC	3.57196	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	2500.00
BLOCK FUEL (LBS)	55262.38
BLOCK TIME (HRS)	5.99
FLIGHT TIME (HRS)	5.66
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4142.22
FLIGHTS PER A/C PER YEAR	691.59
FARE ($)	299.45
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

- -

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR $M	CUMULATIVE DEPRECIATION $M	AVERAGE BOOK VALUE OF FLEET $M	REVENUE $M	INTEREST EXPENSE $M	OPERATING EXPENSE $M	CASH FLOW $M	ROI PERCENT
1	5.0	8.0	515.81	29.01	486.80	246.03	61.90	218.66	-236.60	5.62
2	13.0	8.0	1341.11	104.45	1236.66	639.67	151.03	568.52	-301.55	5.75
3	20.4	7.0	2101.93	222.69	1879.25	1002.57	222.52	891.05	-327.01	5.93
4	23.0	0.0	2372.74	356.15	2016.58	1131.73	226.54	1005.84	-157.01	6.24
5	23.0	0.0	2372.74	489.62	1883.12	1131.73	198.07	1005.84	-128.54	6.69
6	23.0	0.0	2372.74	623.08	1749.65	1131.73	169.60	1005.84	-100.06	7.19
7	23.0	0.0	2372.74	756.55	1616.19	1131.73	141.13	1005.84	-71.59	7.79
8	23.0	0.0	2372.74	890.02	1482.72	1131.73	112.65	1005.84	-49.74	8.04
9	23.0	0.0	2372.74	1023.48	1349.25	1131.73	84.18	1005.84	-35.50	7.78
10	23.0	0.0	2372.74	1156.95	1215.79	1131.73	55.71	1005.84	-21.26	7.47
11	23.0	0.0	2372.74	1290.41	1082.32	1131.73	27.23	1005.84	59.00	7.07
12	23.0	0.0	2372.74	1423.88	948.86	1131.73	8.67	1005.84	134.31	7.09
13	23.0	0.0	2372.74	1557.35	815.39	1131.73	-0.00	1005.84	196.41	7.72
14	23.0	0.0	2372.74	1690.81	681.92	1131.73	-0.00	1005.84	196.41	9.23
15	23.0	0.0	2372.74	1824.28	548.46	1131.73	-0.00	1005.84	196.41	11.48
16	23.0	0.0	2372.74	1957.75	414.99	1131.73	-0.00	1005.84	196.41	15.17

AVG ROI OVER THE 16 YEAR PERIOD = 7.34 PERCENT

IDEA (Unsized)

SUMMARY ID NO. 1 — A S S E T   P A R A M E T R I C   A N A L Y S I S — MAY 24 1984

AIRCRAFT MODEL --CL1633-11-1
I.O.C. DATE --1990
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 751000
SLS SCALE 1.0 = 39000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 25.00 DEG
WING TAPER RATIO = 0.246

1 W/S	131.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.249	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	25.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4600	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	470224	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	152621	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	244103	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	317603	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	1.001	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	39029	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3589.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	207.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	548.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	567.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	15.41	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	6.90	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	201.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.019	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	3.356	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	88.28	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	30.949	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	3.605	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.773	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - 0/0	4.87	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	37000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	115243	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
34 MISN V1(2,1)	41000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 MISN V2(2,1)	55262	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
36 TAKEOFF DST(1)	9136	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(1)	0.1103	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 TAKEOFF DST(2)	9242	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 CLIMB GRAD(2)	0.0452	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
40 CTOL LNDG D(1)	6145	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
41 AP SPEED-KT(1)	135.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
42 CTOL LNDG D(2)	6752	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
43 AP SPEED-KT(2)	144.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
44 CTOL LNDG D(3)	7358	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
45 AP SPEED-KT(3)	153.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
46 SEP(1) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
47 SEP(2) - FPS	5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
48 SEP(3) - FPS	5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
49 SEP(4) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

IDEA (Unsized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

CONFIGURATION GEOMETRY

BASIC WING--	AREA(SQ FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	MAC(FT)		
	3317.5	199.53	0.246	25.000	27.327	18.66		
WING PANELS--	AREA(SQ FT)	EXP. AREA	AVG T/C	L.E. SWEEP	SFLE(SQ FT)	REF L(FT)		
	723.3	32.3	9.29	27.327	0.0	33.40		
	1680.8	1680.8	10.00	27.327	0.0	26.42		
	242.1	242.1	11.16	27.327	0.0	17.34		
	1055.8	1055.8	10.45	27.327	0.0	12.81		
	101.6	101.6	9.16	27.327	0.0	7.30		
TOTAL WING--	AREA(SQ FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	MAC(FT)	L(FT)	E SWP
	3803.5	10.47	10.02	38.00	6.58	23.32	58.13	16.389
FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)			
	201.00	10532.3	19.32	19.58	301.21			
	BW(FT)	BH(FT)	SBW(SQ FT)					
	19.58	19.58	10538.00					
HORZ. TAIL 1--	SHT1(SQ FT)	SHX1(SQ FT)	REF L1(FT)	L HT1(FT)	HT1 VOL COEF			
	487.27	323.56	9.24	78.81	0.6204			
HORZ. TAIL 2--	SHT2(SQ FT)	SHX2(SQ FT)	REF L2(FT)	L HT2(FT)	HT2 VOL COEF			
	0.0	0.0	0.0	201.00	0.0			
VERT. TAIL 1--	SVT1(SQ FT)	SVX1(SQ FT)	REF L1(FT)	L VT1(FT)	VT1 VOL COEF			
	498.58	281.72	15.89	77.78	0.0586			
VERT. TAIL 2--	SVT2(SQ FT)	SVX2(SQ FT)	REF L2(FT)	L VT2(FT)	VT2 VOL COEF			
	0.0	0.0	0.0	201.00	0.0			
PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)	
	14.81	6.64	18.88	7.82	866.26	2.	19.43	
FUEL TANKS--	WING(CU FT)	BOX(CU FT)	FUS(CU FT)					
	2303.38	454.51	999999.00					
WETTED VOLUMES--	BODY	WING	TAILS	PODS	PYLONS	PONTOONS	TOTAL	
	50128.92	4260.76	614.27	1533.64	130.30	0.0	56667.89	

IDEA (Sized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

W E I G H T S T A T E M E N T

	WEIGHT(POUNDS)
GROSS WEIGHT	(434596.)
FUEL AVAILABLE	127572.
EXTERNAL	0.
INTERNAL	127580.
ZERO FUEL WEIGHT	307024.
PAYLOAD	73500.
PASSENGERS	57750.
BAGGAGE	15750.
CARGO	0.
STORES	0.
OPERATIONAL EMPTY WEIGHT	233524.
OPERATIONAL ITEMS	10305.
STANDARD ITEMS	9538.
EMPTY WEIGHT	213682.
STRUCTURE	132674.
WING	48993.
ROTOR	0.
TAIL	3367.
BODY	54144.
ALIGHTING GEAR	17732.
ENGINE SECTION AND NACELLE	8438.
PROPULSION	23842.
CRUISE ENGINES	18385.
LIFT ENGINES	0.
THRUST REVERSER	2685.
EXHAUST SYSTEM	0.
ENGINE CONTROL	93.
STARTING SYSTEM	140.
PROPELLERS	0.
LUBRICATING SYSTEM	0.
FUEL SYSTEM	2539.
DRIVE SYSTEM (POWER TRANS)	0.
SYSTEMS	57165.
FLIGHT CONTROLS	4142.
AUXILIARY POWER PLANT	962.
INSTRUMENTS	1045.
HYDRAULIC AND PNEUMATIC	0.
ELECTRICAL	5403.
AVIONICS	2827.
ARMAMENT	0.
FURNISHINGS AND EQUIPMENT	38504.
AIR CONDITIONING	3848.
ANTI-ICING	434.
PHOTOGRAPHIC	0.
LOAD AND HANDLING	0.

WEIGHT FRACTION	(PERCENT)
FUEL	29.35
PAYLOAD	16.91
OPERATIONAL ITEMS	4.57
STRUCTURE	30.53
PROPULSION	5.49
SYSTEMS	13.15
TOTAL	(100.)

IDEA (Sized)

MISSION SUMMARY

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	434596.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	434596.	533.	533.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	434063.	2241.	2774.	25.	25.	5.6	6.6	0.	751201.	0.	0.617	20.37	0.458
ACCEL	10000.	0.456	431822.	339.	3113.	5.	30.	0.9	7.5	0.	751201.	0.	0.510	19.71	0.484
CLIMB	10000.	0.547	431483.	10415.	13528.	280.	310.	38.1	45.7	0.	751201.	0.	0.461	18.85	0.538
CRUISE	36000.	0.800	421067.	0.	13528.	0.	310.	0.0	45.7	0.	-751101.	0.	0.595	20.07	0.539
ACCEL	36000.	0.800	421067.	0.	13528.	0.	310.	0.0	45.7	0.	751201.	0.	0.595	20.07	0.539
ACCEL	36000.	0.800	421067.	0.	13529.	0.	310.	0.0	45.7	0.	751201.	0.	0.595	20.07	0.539
CRUISE	36000.	0.800	421067.	90308.	103836.	4090.	4400.	534.7	580.3	0.	-751101.	0.	0.597	19.91	0.539
CLIMB	41000.	0.800	330760.	0.	103836.	0.	4400.	0.0	580.3	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	330760.	4.	103840.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.594	19.74	0.541
CLIMB	41000.	0.800	330756.	0.	103840.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	330756.	0.	103840.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.594	19.74	0.541
DESCENT	41000.	0.800	330756.	1054.	104893.	120.	4520.	16.6	597.0	0.	751503.	0.	0.402	17.17	-0.311
DECEL	10000.	0.547	329702.	39.	104933.	5.	4525.	0.9	597.9	0.	751503.	0.	0.394	17.57	-18.187
DESCENT	10000.	0.456	329663.	355.	105288.	31.	4556.	6.8	604.6	0.	751503.	0.	0.469	19.10	6.546
CRUISE	41000.	0.800	329308.	872.	106160.	44.	4600.	5.8	610.4	0.	-751101.	0.	0.590	19.73	0.542
LOITER	1500.	0.295	328436.	405.	106564.	0.	4600.	3.0	613.4	0.	-751101.	0.	0.810	19.94	0.492
CRUISE	1500.	0.378	328031.	310.	106874.	0.	4600.	2.0	615.4	0.	-751101.	0.	0.493	19.44	0.551
RESET	0.	0.0	327722.	0.	106874.	-4600.	0.	0.0	615.4	0.	0.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	327722.	9138.	116011.	0.	0.	61.5	677.0	0.	-751101.	0.	0.580	19.66	0.542
TAKEOFF POWER 1	0.	0.0	318584.	0.	116011.	0.	0.	0.0	677.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	318584.	533.	116544.	0.	0.	1.0	678.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	318051.	1439.	117982.	16.	16.	3.6	681.6	0.	751201.	0.	0.451	18.76	0.458
ACCEL	10000.	0.456	316613.	218.	118201.	3.	19.	0.6	682.2	0.	751201.	0.	0.373	17.04	0.484

IDEA (Sized)

CLIMB	10000.	0.547	316394.	3615.	121816.	81.	100.	11.8	693.9	0.	751201.	0.	0.316	15.45	0.529
CRUISE	30000.	0.695	312779.	0.	121816.	0.	100.	0.0	693.9	0.	-751101.	0.	0.443	18.45	0.535
DESCENT	30000.	0.750	312779.	282.	122097.	53.	153.	7.9	701.9	0.	751503.	0.	0.315	15.41	-1.053
DECEL	10000.	0.547	312498.	38.	122135.	4.	158.	0.8	702.7	0.	751503.	0.	0.374	17.04	-18.276
DESCENT	10000.	0.456	312460.	347.	122482.	30.	188.	6.6	709.3	0.	751503.	0.	0.445	18.64	6.543
CRUISE	30000.	0.695	312112.	270.	122752.	12.	200.	1.8	711.1	0.	-751101.	0.	0.442	18.42	0.535
CRUISE	1500.	0.378	311843.	4840.	127592.	0.	200.	32.0	743.1	0.	-751101.	0.	0.465	19.00	0.557

WTO = 434595.9 FUEL A=127571.6 FUEL R=127591.6

IDEA (Sized)

A L T E R N A T E   M I S S I O N   N O. 1   S U M M A R Y

ATX-350I, AVRG STAGE L = 2500 NMI, M = 0.80, 65% LF

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	349845.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	349845.	533.	533.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	349312.	1625.	2158.	18.	18.	4.1	5.1	0.	751201.	0.	0.496	19.52	0.458
ACCEL	10000.	0.456	347687.	247.	2404.	4.	22.	0.7	5.7	0.	751201.	0.	0.410	17.97	0.484
CLIMB	10000.	0.547	347440.	7878.	10282.	227.	248.	30.8	36.5	0.	751201.	0.	0.411	17.42	0.538
CRUISE	40000.	0.800	339562.	0.	10282.	0.	248.	0.0	36.5	0.	-751101.	0.	0.581	19.74	0.541
ACCEL	40000.	0.800	339562.	0.	10282.	0.	248.	0.0	36.5	0.	751201.	0.	0.581	19.74	0.540
ACCEL	40000.	0.800	339562.	0.	10283.	0.	248.	0.0	36.5	0.	751201.	0.	0.581	19.74	0.540
CRUISE	40000.	0.800	339562.	39438.	49720.	2052.	2300.	268.2	304.7	0.	-751101.	0.	0.580	19.65	0.543
CLIMB	42000.	0.800	300125.	0.	49720.	0.	2300.	0.0	304.7	0.	751201.	0.	0.0	0.0	0.0
CRUISE	42000.	0.800	300125.	0.	49720.	0.	2300.	0.0	304.7	0.	-751101.	0.	0.565	19.49	0.544
CLIMB	42000.	0.800	300124.	0.	49720.	0.	2300.	0.0	304.7	0.	751201.	0.	0.0	0.0	0.0
CRUISE	42000.	0.800	300124.	0.	49720.	0.	2300.	0.0	304.7	0.	-751101.	0.	0.565	19.49	0.544
DESCENT	42000.	0.800	300124.	1222.	50942.	129.	2429.	17.7	322.4	0.	751503.	0.	0.375	16.41	-0.176
DECEL	10000.	0.547	298903.	37.	50979.	4.	2433.	0.8	323.2	0.	751503.	0.	0.358	16.59	-18.343
DESCENT	10000.	0.456	298865.	340.	51319.	29.	2462.	6.5	329.7	0.	751503.	0.	0.426	18.23	6.541
CRUISE	42000.	0.800	298526.	684.	52003.	38.	2500.	4.9	334.6	0.	-751101.	0.	0.562	19.46	0.545
LOITER	1500.	0.280	297842.	370.	52373.	0.	2500.	3.0	337.6	0.	-751101.	0.	0.816	19.86	0.494
CRUISE	1500.	0.378	297472.	298.	52671.	0.	2500.	2.0	339.6	0.	-751101.	0.	0.447	18.65	0.561
RESET	0.	0.0	297174.	0.	52671.	-2500.	0.	0.0	339.6	0.	0.	0.	0.0	0.0	0.0
CRUISE	42001.	0.800	297174.	4686.	57357.	0.	0.	34.0	373.5	0.	-751101.	0.	0.555	19.40	0.545
TAKEOFF POWER 1	0.	0.0	292487.	0.	57357.	0.	0.	0.0	373.5	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	292487.	533.	57890.	0.	0.	1.0	374.5	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	291954.	1294.	59184.	15.	15.	3.2	377.8	0.	751201.	0.	0.413	17.93	0.458
ACCEL	10000.	0.456	290660.	196.	59381.	3.	17.	0.5	378.3	0.	751201.	0.	0.343	16.15	0.484

IDEA (Sized)

CLIMB	10000.	0.547	290463.	3373.	62754.	76.	94.	11.1	389.4	0.	751201.	0.	0.293	14.60	0.529
CRUISE	30000.	0.660	287090.	132.	62886.	6.	100.	1.0	390.4	0.	-751101.	0.	0.451	18.48	0.528
DESCENT	30000.	0.750	286958.	264.	63150.	50.	150.	7.4	397.8	0.	751503.	0.	0.289	14.47	-1.052
DECEL	10000.	0.547	286694.	36.	63186.	4.	154.	0.8	398.6	0.	751503.	0.	0.343	16.16	-18.420
DESCENT	10000.	0.456	286658.	333.	63519.	29.	183.	6.3	404.9	0.	751503.	0.	0.408	17.83	6.538
CRUISE	30000.	0.660	286325.	359.	63878.	17.	200.	2.6	407.6	0.	-751101.	0.	0.449	18.45	0.528
CRUISE	1500.	0.375	285966.	4651.	68529.	0.	200.	32.0	439.6	0.	-751101.	0.	0.433	18.35	0.564

IDEA (Sized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
FLIGHT CREW	0.50914	14.82321
FUEL AND OIL	1.48904	43.35191
INSURANCE	0.04833	1.40694
DEPRECIATION	0.76202	22.18538
MAINTENANCE	0.62624	18.23251
TOTAL DOC	3.43477	100.000

INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.29589	10.78419
AIRCRAFT CONTROL	0.00854	0.31136
CABIN ATTENDANT	0.50658	18.46315
FOOD AND BEVERAGE	0.18145	6.61335
PASSENGER HANDLING	0.41303	15.05375
CARGO HANDLING	0.14903	5.43167
OTHER PASSENGER EXPENSE	0.99450	36.24640
OTHER CARGO EXPENSE	0.02268	0.82670
GENERAL + ADMINISTRATION	0.17202	6.26951
TOTAL IOC	2.74372	100.000

MISC. DATA	
FLIGHT DISTANCE (N. MI.)	4600.00
BLOCK FUEL (LBS)	107519.88
BLOCK TIME (HRS)	10.59
FLIGHT TIME (HRS)	10.26
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4618.64
FLIGHTS PER A/C PER YEAR	436.10
FARE ($)	462.19
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

- -

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	495.44	27.87	467.57	241.41	59.45	216.90	-229.03	5.24
2	13.0	8.0	1288.14	100.33	1187.81	627.66	145.06	563.95	-294.26	5.36
3	20.4	7.0	2018.91	213.89	1805.02	983.74	213.73	883.88	-321.35	5.53
4	23.0	0.0	2279.01	342.08	1936.93	1110.48	217.60	997.76	-159.00	5.82
5	23.0	0.0	2279.01	470.28	1808.73	1110.48	190.25	997.76	-131.65	6.23
6	23.0	0.0	2279.01	598.47	1680.54	1110.48	162.90	997.76	-104.30	6.71
7	23.0	0.0	2279.01	726.67	1552.35	1110.48	135.55	997.76	-76.95	7.26
8	23.0	0.0	2279.01	854.86	1424.15	1110.48	108.20	997.76	-51.87	7.76
9	23.0	0.0	2279.01	983.05	1295.96	1110.48	80.86	997.76	-38.19	7.47
10	23.0	0.0	2279.01	1111.25	1167.76	1110.48	53.51	997.76	-24.52	7.12
11	23.0	0.0	2279.01	1239.44	1039.57	1110.48	26.16	997.76	52.57	6.68
12	23.0	0.0	2279.01	1367.64	911.38	1110.48	8.32	997.76	124.91	6.64
13	23.0	0.0	2279.01	1495.83	783.18	1110.48	-0.00	997.76	134.56	7.20
14	23.0	0.0	2279.01	1624.02	654.99	1110.48	-0.00	997.76	184.56	8.61
15	23.0	0.0	2279.01	1752.22	526.79	1110.48	-0.00	997.76	184.56	10.70
16	23.0	0.0	2279.01	1880.41	398.60	1110.48	-0.00	997.76	184.56	14.14

AVG ROI OVER THE 16 YEAR PERIOD = 6.89 PERCENT

IDEA (Sized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.53013	14.99726
FUEL AND OIL	1.35870	38.43716
INSURANCE	0.05610	1.58699
DEPRECIATION	0.88458	25.02452
MAINTENANCE	0.70535	19.95407
TOTAL DOC	3.53485	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.54443	15.44930
AIRCRAFT CONTROL	0.01572	0.44605
CABIN ATTENDANT	0.52746	14.96762
FOOD AND BEVERAGE	0.18893	5.36127
PASSENGER HANDLING	0.75998	21.56581
CARGO HANDLING	0.27421	7.78134
OTHER PASSENGER EXPENSE	0.99450	28.22075
OTHER CARGO EXPENSE	0.02268	0.64365
GENERAL + ADMINISTRATION	0.19608	5.56425
TOTAL IOC	3.52400	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	2500.00
BLOCK FUEL (LBS)	53317.05
BLOCK TIME (HRS)	5.99
FLIGHT TIME (HRS)	5.66
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4142.71
FLIGHTS PER A/C PER YEAR	691.25
FARE ($)	299.45
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	495.44	27.87	467.57	245.91	59.45	213.47	-221.11	6.94
2	13.0	8.0	1288.14	100.33	1187.81	639.35	145.06	555.03	-273.66	7.10
3	20.4	7.0	2018.91	213.89	1805.02	1002.07	213.73	869.91	-289.05	7.32
4	23.0	0.0	2279.01	342.08	1936.93	1131.17	217.60	981.98	-122.54	7.70
5	23.0	0.0	2279.01	470.28	1808.73	1131.17	190.25	981.98	-95.19	8.25
6	23.0	0.0	2279.01	598.47	1680.54	1131.17	162.90	981.98	-67.84	8.88
7	23.0	0.0	2279.01	726.67	1552.35	1131.17	135.55	981.98	-47.31	9.17
8	23.0	0.0	2279.01	854.86	1424.15	1131.17	108.20	981.98	-33.64	9.04
9	23.0	0.0	2279.01	983.05	1295.96	1131.17	80.86	981.98	-19.96	8.88
10	23.0	0.0	2279.01	1111.25	1167.76	1131.17	53.51	981.98	-6.29	8.68
11	23.0	0.0	2279.01	1239.44	1039.57	1131.17	26.16	981.98	70.80	8.43
12	23.0	0.0	2279.01	1367.64	911.38	1131.17	8.32	981.98	143.14	8.64
13	23.0	0.0	2279.01	1495.83	783.18	1131.17	-0.00	981.98	202.79	9.52
14	23.0	0.0	2279.01	1624.02	654.99	1131.17	-0.00	981.98	202.79	11.39
15	23.0	0.0	2279.01	1752.22	526.79	1131.17	-0.00	981.98	202.79	14.16
16	23.0	0.0	2279.01	1880.41	398.60	1131.17	-0.00	981.98	202.79	18.71

AVG ROI OVER THE 16 YEAR PERIOD = 8.84 PERCENT

IDEA (Sized)

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S MAY 24 1984

AIRCRAFT MODEL --CL1633-11-1 ENGINE I.D. -- 751000 WING QUARTER CHORD SWEEP = 25.00 DEG
I.O.C. DATE --1990 SLS SCALE 1.0 = 39000 WING TAPER RATIO = 0.246
DESIGN SPEED --SUBSONIC NUMBER OF ENGINES = 3.

1 W/S	131.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.249	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	25.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4600	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	434596	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	127572	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	233524	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	307024	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	0.925	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	36071	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3318.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	199.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	487.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	498.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	14.81	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	6.64	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	201.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.082	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	3.204	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	84.83	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	29.726	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	3.435	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.744	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - 0/0	6.89	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	36000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	107520	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
34 MISN V1(2,1)	40000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 MISN V2(2,1)	53317	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
36 TAKEOFF DST(1)	9166	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(1)	0.1101	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 TAKEOFF DST(2)	9282	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 CLIMB GRAD(2)	0.0450	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
40 CTOL LNDG D(1)	6128	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
41 AP SPEED-KT(1)	135.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
42 CTOL LNDG D(2)	6784	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
43 AP SPEED-KT(2)	145.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
44 CTOL LNDG D(3)	7439	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
45 AP SPEED-KT(3)	154.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
46 SEP(1) - FPS	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
47 SEP(2) - FPS	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
48 SEP(3) - FPS	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
49 SEP(4) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

IDEA (Sized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

C O N F I G U R A T I O N   G E O M E T R Y

BASIC WING--	AREA(SQ FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	MAC(FT)		
	3589.5	207.54	0.246	25.000	27.327	19.41		
WING PANELS--	AREA(SQ FT)	EXP. AREA	AVG T/C	L.E. SWEEP	SFLE(SQ FT)	REF L(FT)		
	782.5	62.1	9.29	27.327	0.0	34.83		
	1818.6	1818.6	10.00	27.327	0.0	27.48		
	261.9	261.9	11.16	27.327	0.0	18.04		
	1142.3	1142.3	10.45	27.327	0.0	13.33		
	110.0	110.0	9.16	27.327	0.0	7.59		
TOTAL WING--	AREA(SQ FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	MAC(FT)	L(FT)	E SWP
	4115.3	10.47	10.02	39.53	6.84	24.25	60.46	16.389
FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWW(FT)	EQUIV D(FT)	SPI(SQ FT)			
	201.00	10603.5	19.32	19.58	301.21			
	BW(FT)	BH(FT)	SBW(SQ FT)					
	19.58	19.58	10538.00					
HORZ. TAIL 1--	SHT1(SQ FT)	SHX1(SQ FT)	REF L1(FT)	L HT1(FT)	HT1 VOL COEF			
	548.40	373.81	9.90	78.81	0.6204			
HORZ. TAIL 2--	SHT2(SQ FT)	SHX2(SQ FT)	REF L2(FT)	L HT2(FT)	HT2 VOL COEF			
	0.0	0.0	0.0	201.00	0.0			
VERT. TAIL 1--	SVT1(SQ FT)	SVX1(SQ FT)	REF L1(FT)	L VT1(FT)	VT1 VOL COEF			
	567.46	334.28	17.21	76.91	0.0586			
VERT. TAIL 2--	SVT2(SQ FT)	SVX2(SQ FT)	REF L2(FT)	L VT2(FT)	VT2 VOL COEF			
	0.0	0.0	0.0	201.00	0.0			
PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)	
	15.41	6.90	19.64	8.13	937.28	2.	20.21	
FUEL TANKS--	WING(CU FT)	BOX(CU FT)	FUS(CU FT)					
	2612.07	496.44	999999.00					
WETTED VOLUMES--	BODY	WING	TAILS	PODS	PYLONS	PONTOONS	TOTAL	
	50311.41	4853.41	777.55	1726.05	146.65	0.0	57815.07	

Alt IDEA (Unsized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

WEIGHT STATEMENT

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(470224.)		
FUEL AVAILABLE	160545.	FUEL	34.14
EXTERNAL	0.		
INTERNAL	160545.		
ZERO FUEL WEIGHT	309679.		
PAYLOAD	73500.	PAYLOAD	15.63
PASSENGERS	57750.		
BAGGAGE	15750.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	236179.		
OPERATIONAL ITEMS	10319.	OPERATIONAL ITEMS	4.23
STANDARD ITEMS	9579.		
EMPTY WEIGHT	216280.		
STRUCTURE	133179.	STRUCTURE	28.32
WING	46616.		
ROTOR	0.		
TAIL	3805.		
BODY	54923.		
ALIGHTING GEAR	18602.		
ENGINE SECTION AND NACELLE	9233.		
PROPULSION	26070.	PROPULSION	5.54
CRUISE ENGINES	20129.		
LIFT ENGINES	0.		
THRUST REVERSER	2831.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	100.		
STARTING SYSTEM	140.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2869.		
DRIVE SYSTEM (POWER TRANS)	0.		
SYSTEMS	57032.		
FLIGHT CONTROLS	4328.		
AUXILIARY POWER PLANT	962.		
INSTRUMENTS	1058.		
HYDRAULIC AND PNEUMATIC	0.		
ELECTRICAL	5348.		
AVIONICS	2827.	SYSTEMS	12.13
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	38504.		
AIR CONDITIONING	3557.		
ANTI-ICING	448.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

Alt IDEA (Unsized)

MISSION SUMMARY

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	470224.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	470224.	576.	576.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	469648.	2408.	2984.	25.	25.	5.6	6.6	0.	751201.	0.	0.617	20.59	0.458
ACCEL	10000.	0.456	467240.	362.	3347.	5.	30.	0.9	7.5	0.	751201.	0.	0.510	19.95	0.484
CLIMB	10000.	0.547	466877.	11706.	15052.	298.	328.	40.5	48.0	0.	751201.	0.	0.476	19.22	0.538
CRUISE	37000.	0.800	455172.	0.	15052.	0.	328.	0.0	48.0	0.	-751101.	0.	0.624	20.30	0.538
ACCEL	37000.	0.800	455172.	0.	15052.	0.	328.	0.0	48.0	0.	751201.	0.	0.624	20.30	0.538
ACCEL	37000.	0.800	455172.	0.	15052.	0.	328.	0.0	48.0	0.	751201.	0.	0.624	20.30	0.538
CRUISE	37000.	0.800	455171.	96257.	111309.	4072.	4400.	532.3	580.4	0.	-751101.	0.	0.607	20.15	0.539
CLIMB	41000.	0.800	358915.	0.	111309.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	358915.	0.	111309.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.596	19.98	0.542
CLIMB	41000.	0.800	358914.	0.	111309.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	358914.	0.	111309.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.596	19.98	0.542
DESCENT	41000.	0.800	358914.	1086.	112396.	119.	4519.	16.5	596.9	0.	751503.	0.	0.402	17.42	-0.338
DECEL	10000.	0.547	357828.	43.	112439.	5.	4524.	0.9	597.8	0.	751503.	0.	0.395	17.86	-18.175
DESCENT	10000.	0.456	357785.	390.	112829.	31.	4555.	6.9	604.6	0.	751503.	0.	0.471	19.38	6.545
CRUISE	41000.	0.800	357395.	946.	113775.	45.	4600.	5.9	610.5	0.	-751101.	0.	0.592	19.96	0.542
LOITER	1500.	0.295	356449.	436.	114211.	0.	4600.	3.0	613.5	0.	-751101.	0.	0.813	20.10	0.492
CRUISE	1500.	0.378	356013.	333.	114543.	0.	4600.	2.0	615.5	0.	-751101.	0.	0.495	19.71	0.553
RESET	0.	0.0	355680.	0.	114543.	-4600.	0.	0.0	615.5	0.	0.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	355680.	9807.	124350.	0.	0.	61.5	677.0	0.	-751101.	0.	0.582	19.90	0.542
TAKEOFF POWER 1	0.	0.0	345873.	0.	124350.	0.	0.	0.0	677.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	345873.	576.	124926.	0.	0.	1.0	678.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	345297.	1553.	126480.	16.	16.	3.6	681.6	0.	751201.	0.	0.452	19.04	0.458
ACCEL	10000.	0.456	343744.	235.	126715.	3.	19.	0.6	682.2	0.	751201.	0.	0.375	17.33	0.484

Alt IDEA (Unsized)

CLIMB	10000.	0.547	343508.	3900.	130614.	81.	100.	11.8	694.0	0.	751201.	0.	0.318	15.75	0.529
CRUISE	30000.	0.700	339609.	0.	130614.	0.	100.	0.0	694.0	0.	-751101.	0.	0.438	18.62	0.538
DESCENT	30000.	0.750	339609.	310.	130924.	54.	154.	8.1	702.0	0.	751503.	0.	0.316	15.69	-1.053
DECEL	10000.	0.547	339299.	42.	130966.	5.	159.	0.8	702.9	0.	751503.	0.	0.375	17.33	-18.266
DESCENT	10000.	0.456	339257.	382.	131348.	30.	189.	6.7	709.6	0.	751503.	0.	0.446	18.93	6.542
CRUISE	30000.	0.695	338875.	256.	131603.	11.	200.	1.6	711.2	0.	-751101.	0.	0.443	18.71	0.536
CRUISE	1500.	0.378	338619.	5197.	136801.	0.	200.	32.0	743.2	0.	-751101.	0.	0.467	19.28	0.559

WTO = 470224.0 FUEL A=160545.4 FUEL R=136800.6

Alt IDEA (Unsized)

ALTERNATE MISSION NO. 1 SUMMARY

ATX-350I, AVRG STAGE L = 2500 NMI, M = 0.80, 65% LF

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	353798.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	353798.	576.	576.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	353222.	1599.	2175.	17.	17.	3.7	4.7	0.	751201.	0.	0.463	19.24	0.458
ACCEL	10000.	0.456	351623.	242.	2417.	3.	20.	0.6	5.3	0.	751201.	0.	0.383	17.56	0.484
CLIMB	10000.	0.547	351381.	7625.	10042.	206.	226.	28.0	33.3	0.	751201.	0.	0.395	17.17	0.538
CRUISE	41000.	0.800	343756.	0.	10042.	0.	226.	0.0	33.3	0.	-751101.	0.	0.570	19.83	0.543
ACCEL	41000.	0.800	343756.	0.	10042.	0.	226.	0.0	33.3	0.	751201.	0.	0.570	19.83	0.541
ACCEL	41000.	0.800	343756.	0.	10042.	0.	226.	0.0	33.3	0.	751201.	0.	0.570	19.83	0.541
CRUISE	41000.	0.800	343756.	40359.	50402.	2074.	2300.	271.1	304.4	0.	-751101.	0.	0.570	19.73	0.545
CLIMB	43000.	0.800	303396.	0.	50402.	0.	2300.	0.0	304.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	43000.	0.800	303396.	4.	50406.	0.	2300.	0.0	304.4	0.	-751101.	0.	0.554	19.55	0.548
CLIMB	43000.	0.800	303392.	0.	50406.	0.	2300.	0.0	304.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	43000.	0.800	303392.	0.	50406.	0.	2300.	0.0	304.4	0.	-751101.	0.	0.554	19.55	0.548
DESCENT	43000.	0.800	303392.	1444.	51850.	137.	2437.	18.7	323.1	0.	751503.	0.	0.359	16.10	-0.091
DECEL	10000.	0.547	301948.	39.	51889.	4.	2441.	0.8	323.9	0.	751503.	0.	0.334	16.11	-18.463
DESCENT	10000.	0.456	301909.	360.	52250.	29.	2470.	6.4	330.2	0.	751503.	0.	0.397	17.83	6.535
CRUISE	43000.	0.800	301548.	558.	52808.	30.	2500.	4.0	334.2	0.	-751101.	0.	0.550	19.52	0.548
LOITER	1500.	0.250	300990.	372.	53180.	0.	2500.	3.0	337.2	0.	-751101.	0.	0.956	19.19	0.475
CRUISE	1500.	0.378	300618.	312.	53492.	0.	2500.	2.0	339.2	0.	-751101.	0.	0.418	18.26	0.570
RESET	0.	0.0	300305.	0.	53492.	-2500.	0.	0.0	339.2	0.	0.	0.	0.0	0.0	0.0
CRUISE	42001.	0.800	300305.	4793.	58285.	0.	0.	33.9	373.1	0.	-751101.	0.	0.519	19.23	0.547
TAKEOFF POWER 1	0.	0.0	295512.	0.	58285.	0.	0.	0.0	373.1	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	295512.	576.	58862.	0.	0.	1.0	374.1	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	294936.	1282.	60143.	13.	13.	3.0	377.1	0.	751201.	0.	0.385	17.51	0.458
ACCEL	10000.	0.456	293654.	194.	60337.	3.	16.	0.5	377.6	0.	751201.	0.	0.320	15.66	0.484

Alt IDEA (Unsized)

CLIMB	10000.	0.547	293460.	3266.	63604.	68.	84.	9.9	387.5	0.	751201.	0.	0.274	14.07	0.529
CRUISE	30000.	0.645	290194.	345.	63949.	16.	100.	2.5	390.0	0.	-751101.	0.	0.441	18.48	0.526
DESCENT	30000.	0.750	289849.	274.	64223.	48.	148.	7.1	397.1	0.	751503.	0.	0.270	13.94	-1.052
DECEL	10000.	0.547	289575.	38.	64260.	4.	152.	0.8	397.9	0.	751503.	0.	0.320	15.66	-18.532
DESCENT	10000.	0.456	289537.	352.	64613.	28.	180.	6.2	404.1	0.	751503.	0.	0.381	17.42	6.533
CRUISE	30000.	0.645	289185.	431.	65044.	20.	200.	3.1	407.2	0.	-751101.	0.	0.442	18.51	0.526
CRUISE	1500.	0.375	288754.	4875.	69919.	0.	200.	32.0	439.2	0.	-751101.	0.	0.404	17.95	0.573

Alt IDEA (Unsized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.50919	14.24176
FUEL AND OIL	1.59595	44.63786
INSURANCE	0.04999	1.39818
DEPRECIATION	0.79084	22.11946
MAINTENANCE	0.62936	17.60278
TOTAL DOC	3.57533	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.31833	11.48951
AIRCRAFT CONTROL	0.00854	0.30834
CABIN ATTENDANT	0.50662	18.28552
FOOD AND BEVERAGE	0.18147	6.54972
PASSENGER HANDLING	0.41303	14.90757
CARGO HANDLING	0.14903	5.37892
OTHER PASSENGER EXPENSE	0.99450	35.89441
OTHER CARGO EXPENSE	0.02268	0.81867
GENERAL + ADMINISTRATION	0.17642	6.36755
TOTAL IOC	2.77062	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	4600.00
BLOCK FUEL (LBS)	115242.50
BLOCK TIME (HRS)	10.59
FLIGHT TIME (HRS)	10.26
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4618.72
FLIGHTS PER A/C PER YEAR	436.07
FARE ($)	462.19
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	514.06	28.92	485.14	241.39	61.69	222.77	-244.44	3.84
2	13.0	8.0	1336.55	104.10	1232.46	627.62	150.52	579.19	-323.01	3.93
3	20.4	7.0	2094.79	221.93	1872.86	983.67	221.76	907.77	-361.15	4.05
4	23.0	0.0	2364.67	354.94	2009.73	1110.40	225.77	1024.73	-196.26	4.26
5	23.0	0.0	2364.67	487.95	1876.72	1110.40	197.40	1024.73	-167.88	4.57
6	23.0	0.0	2364.67	620.97	1743.70	1110.40	169.02	1024.73	-139.51	4.91
7	23.0	0.0	2364.67	753.98	1610.69	1110.40	140.65	1024.73	-111.13	5.32
8	23.0	0.0	2364.67	886.99	1477.68	1110.40	112.27	1024.73	-82.76	5.80
9	23.0	0.0	2364.67	1020.00	1344.67	1110.40	83.89	1024.73	-55.27	6.31
10	23.0	0.0	2364.67	1153.01	1211.66	1110.40	55.52	1024.73	-41.08	5.83
11	23.0	0.0	2364.67	1286.03	1078.64	1110.40	27.14	1024.73	38.90	5.23
12	23.0	0.0	2364.67	1419.04	945.63	1110.40	8.64	1024.73	113.96	4.99
13	23.0	0.0	2364.67	1552.05	812.62	1110.40	-0.00	1024.73	175.85	5.27
14	23.0	0.0	2364.67	1685.06	679.61	1110.40	-0.00	1024.73	175.85	6.30
15	23.0	0.0	2364.67	1818.08	546.59	1110.40	-0.00	1024.73	175.85	7.84
16	23.0	0.0	2364.67	1951.09	413.58	1110.40	-0.00	1024.73	175.85	10.36

AVG ROI OVER THE 16 YEAR PERIOD = 5.18 PERCENT

Alt IDEA (Unsized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.52957	14.73503
FUEL AND OIL	1.38097	38.42470
INSURANCE	0.05798	1.61315
DEPRECIATION	0.91720	25.52036
MAINTENANCE	0.70826	19.70673
TOTAL DOC	3.59398	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.58573	16.42197
AIRCRAFT CONTROL	0.01572	0.44071
CABIN ATTENDANT	0.52691	14.77272
FOOD AND BEVERAGE	0.18873	5.29146
PASSENGER HANDLING	0.75998	21.30737
CARGO HANDLING	0.27421	7.68809
OTHER PASSENGER EXPENSE	0.99450	27.88255
OTHER CARGO EXPENSE	0.02268	0.63594
GENERAL + ADMINISTRATION	0.19828	5.55924
TOTAL IOC	3.56674	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	2500.00
BLOCK FUEL (LBS)	54191.67
BLOCK TIME (HRS)	5.99
FLIGHT TIME (HRS)	5.65
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4141.88
FLIGHTS PER A/C PER YEAR	691.84
FARE ($)	299.45
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	514.06	28.92	485.14	246.12	61.69	216.74	-233.69	-6.06
2	13.0	8.0	1336.55	104.10	1232.46	639.90	150.52	563.52	-295.06	6.20
3	20.4	7.0	2094.79	221.93	1872.86	1002.92	221.76	883.21	-317.34	6.39
4	23.0	0.0	2364.67	354.94	2009.73	1132.13	225.77	997.00	-146.81	6.72
5	23.0	0.0	2364.67	487.95	1876.72	1132.13	197.40	997.00	-118.43	7.20
6	23.0	0.0	2364.67	620.97	1743.70	1132.13	169.02	997.00	-90.06	7.75
7	23.0	0.0	2364.67	753.98	1610.69	1132.13	140.65	997.00	-61.68	8.39
8	23.0	0.0	2364.67	886.99	1477.68	1132.13	112.27	997.00	-44.73	8.37
9	23.0	0.0	2364.67	1020.00	1344.67	1132.13	83.89	997.00	-30.54	8.14
10	23.0	0.0	2364.67	1153.01	1211.66	1132.13	55.52	997.00	-16.36	7.87
11	23.0	0.0	2364.67	1286.03	1078.64	1132.13	27.14	997.00	63.63	7.52
12	23.0	0.0	2364.67	1419.04	945.63	1132.13	8.64	997.00	138.68	7.60
13	23.0	0.0	2364.67	1552.05	812.62	1132.13	-0.00	997.00	200.58	8.31
14	23.0	0.0	2364.67	1685.06	679.61	1132.13	-0.00	997.00	200.58	9.94
15	23.0	0.0	2364.67	1818.08	546.59	1132.13	-0.00	997.00	200.58	12.36
16	23.0	0.0	2364.67	1951.09	413.58	1132.13	-0.00	997.00	200.58	16.34

AVG ROI OVER THE 16 YEAR PERIOD = 7.85 PERCENT

Alt IDEA (Unsized)

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S MAY 25 1984

AIRCRAFT MODEL --CL1633-11-1
I.O.C. DATE --1990
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 751000
SLS SCALE 1.0 = 39000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 25.00 DEG
WING TAPER RATIO = 0.246

1 W/S	131.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.249	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	25.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4600	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	470224	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	160545	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	236179	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	309679	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	1.001	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	39029	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3589.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	207.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	548.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	567.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	15.41	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	6.90	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	201.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	0.969	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	3.250	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	87.74	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	30.844	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	3.575	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.771	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - O/O	5.18	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	37000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	115243	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
34 MISN V1(2,1)	41000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 MISN V2(2,1)	54192	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
36 TAKEOFF DST(1)	9136	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(1)	0.1103	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 TAKEOFF DST(2)	9242	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 CLIMB GRAD(2)	0.0452	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
40 CTOL LNDG D(1)	6145	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
41 AP SPEED-KT(1)	135.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
42 CTOL LNDG D(2)	6752	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
43 AP SPEED-KT(2)	144.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
44 CTOL LNDG D(3)	7358	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
45 AP SPEED-KT(3)	153.5	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
46 SEP(1) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
47 SEP(2) - FPS	5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
48 SEP(3) - FPS	5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
49 SEP(4) - FPS	3	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Alt IDEA (Unsized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

CONFIGURATION GEOMETRY

BASIC WING--	AREA(SQ FT)	SPAN(FT)	TAPER RATIO	C/4 SWEEP	L.E. SWEEP	MAC(FT)			
	3195.8	195.83	0.246	25.000	27.327	18.31			
WING PANELS--	AREA(SQ FT)	EXP. AREA	AVG T/C	L.E. SWEEP	SFLE(SQ FT)	REF L(FT)			
	696.7	19.4	9.30	27.327	0.0	32.74			
	1619.1	1619.1	10.00	27.327	0.0	25.93			
	233.2	233.2	11.16	27.327	0.0	17.02			
	1017.0	1017.0	10.45	27.327	0.0	12.58			
	97.9	97.9	9.16	27.327	0.0	7.17			
TOTAL WING--	AREA(SQ FT)	EFF AR	AVG T/C	CR(FT)	CT(FT)	MAC(FT)	L(FT)	E SWP	
	3663.9	10.47	10.02	37.30	6.45	22.88	57.05	16.389	
FUSELAGE--	LENGTH(FT)	S WET(SQ FT)	BWH(FT)	EQUIV D(FT)	SPI(SQ FT)				
	201.00	10500.4	19.32	19.58	301.21				
	BW(FT)	BH(FT)	SBW(SQ FT)						
	19.58	19.58	10538.00						
HORZ. TAIL 1--	SHT1(SQ FT)	SHX1(SQ FT)	REF L1(FT)	L HT1(FT)	HT1 VOL COEF				
	460.69	301.92	8.94	78.81	0.6204				
HORZ. TAIL 2--	SHT2(SQ FT)	SHX2(SQ FT)	REF L2(FT)	L HT2(FT)	HT2 VOL COEF				
	0.0	0.0	0.0	201.00	0.0				
VERT. TAIL 1--	SVT1(SQ FT)	SVX1(SQ FT)	REF L1(FT)	L VT1(FT)	VT1 VOL COEF				
	468.99	259.48	15.30	78.17	0.0586				
VERT. TAIL 2--	SVT2(SQ FT)	SVX2(SQ FT)	REF L2(FT)	L VT2(FT)	VT2 VOL COEF				
	0.0	0.0	0.0	201.00	0.0				
PROPULSION--	ENG L(FT)	ENG D(FT)	POD L(FT)	POD D(FT)	POD S WET	NO. PODS	INLET L(FT)		
	14.54	6.51	18.53	7.67	834.47	2.	19.07		
FUEL TANKS--	WING(CU FT)	BOX(CU FT)	FUS(CU FT)						
	2169.69	435.80	999999.00						
WETTED VOLUMES--	BODY	WING	TAILS	PODS	PYLONS	PONTOONS	TOTAL		
	50049.57	4004.66	548.69	1449.98	123.20	0.0	56176.09		

Alt IDEA (Sized)

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

T/C=10.00 AR=12.00 W/S=131.00 T/W=0.249

W E I G H T   S T A T E M E N T

	WEIGHT(POUNDS)	WEIGHT FRACTION	(PERCENT)
GROSS WEIGHT	(418644.)		
FUEL AVAILABLE	123537.	FUEL	29.51
EXTERNAL	0.		
INTERNAL	123542.		
ZERO FUEL WEIGHT	295107.		
PAYLOAD	73500.	PAYLOAD	17.56
PASSENGERS	57750.		
BAGGAGE	15750.		
CARGO	0.		
STORES	0.		
OPERATIONAL EMPTY WEIGHT	221607.		
OPERATIONAL ITEMS	10298.	OPERATIONAL ITEMS	4.74
STANDARD ITEMS	9533.		
EMPTY WEIGHT	201776.		
STRUCTURE	122225.	STRUCTURE	29.20
WING	40535.		
ROTOR	0.		
TAIL	3177.		
BODY	53868.		
ALIGHTING GEAR	16561.		
ENGINE SECTION AND NACELLE	8084.		
PROPULSION	22993.	PROPULSION	5.49
CRUISE ENGINES	17612.		
LIFT ENGINES	0.		
THRUST REVERSER	2618.		
EXHAUST SYSTEM	0.		
ENGINE CONTROL	89.		
STARTING SYSTEM	140.		
PROPELLERS	0.		
LUBRICATING SYSTEM	0.		
FUEL SYSTEM	2534.		
DRIVE SYSTEM (POWER TRANS)	0.		
SYSTEMS	56558.		
FLIGHT CONTROLS	3957.		
AUXILIARY POWER PLANT	962.		
INSTRUMENTS	1039.		
HYDRAULIC AND PNEUMATIC	0.		
ELECTRICAL	5285.		
AVIONICS	2827.	SYSTEMS	13.51
ARMAMENT	0.		
FURNISHINGS AND EQUIPMENT	38504.		
AIR CONDITIONING	3557.		
ANTI-ICING	427.		
PHOTOGRAPHIC	0.		
LOAD AND HANDLING	0.		
		TOTAL	(100.)

Alt IDEA (Sized)

MISSION SUMMARY

ATX-350I, DESIGN RANGE = 4600 NMI, M = 0.80, INTERN. RESRVS

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	418644.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	418644.	513.	513.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	418131.	2167.	2680.	25.	25.	5.6	6.6	0.	751201.	0.	0.617	20.26	0.458
ACCEL	10000.	0.456	415964.	328.	3009.	5.	30.	0.9	7.6	0.	751201.	0.	0.509	19.58	0.484
CLIMB	10000.	0.547	415635.	10271.	13279.	287.	318.	39.1	46.7	0.	751201.	0.	0.461	18.73	0.538
CRUISE	36000.	0.800	405365.	0.	13279.	0.	318.	0.0	46.7	0.	-751101.	0.	0.595	19.96	0.539
ACCEL	36000.	0.800	405365.	0.	13279.	0.	318.	0.0	46.7	0.	751201.	0.	0.595	19.96	0.539
ACCEL	36000.	0.800	405365.	0.	13279.	0.	318.	0.0	46.7	0.	751201.	0.	0.595	19.96	0.539
CRUISE	36000.	0.800	405365.	87256.	100536.	4082.	4400.	533.7	580.4	0.	-751101.	0.	0.595	19.79	0.539
CLIMB	41000.	0.800	318108.	0.	100536.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	318108.	0.	100536.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.593	19.63	0.541
CLIMB	41000.	0.800	318108.	0.	100536.	0.	4400.	0.0	580.4	0.	751201.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	318108.	0.	100536.	0.	4400.	0.0	580.4	0.	-751101.	0.	0.593	19.63	0.541
DESCENT	41000.	0.800	318108.	1040.	101576.	121.	4521.	16.7	597.1	0.	751503.	0.	0.402	17.04	-0.297
DECEL	10000.	0.547	317068.	38.	101613.	5.	4525.	0.9	597.9	0.	751503.	0.	0.393	17.43	-18.194
DESCENT	10000.	0.456	317031.	339.	101953.	30.	4556.	6.7	604.7	0.	751503.	0.	0.469	18.96	6.547
CRUISE	41000.	0.800	316691.	839.	102791.	44.	4600.	5.8	610.4	0.	-751101.	0.	0.589	19.61	0.541
LOITER	1500.	0.295	315852.	390.	103181.	0.	4600.	3.0	613.4	0.	-751101.	0.	0.809	19.86	0.491
CRUISE	1500.	0.378	315462.	299.	103481.	0.	4600.	2.0	615.4	0.	-751101.	0.	0.492	19.31	0.550
RESET	0.	0.0	315163.	0.	103481.	-4600.	0.	0.0	615.4	0.	0.	0.	0.0	0.0	0.0
CRUISE	41000.	0.800	315163.	8838.	112319.	0.	0.	61.5	677.0	0.	-751101.	0.	0.579	19.54	0.542
TAKEOFF POWER 1	0.	0.0	306324.	0.	112319.	0.	0.	0.0	677.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	306324.	513.	112832.	0.	0.	1.0	678.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	305811.	1387.	114219.	16.	16.	3.6	681.6	0.	751201.	0.	0.450	18.61	0.458
ACCEL	10000.	0.456	304424.	211.	114430.	3.	19.	0.6	682.2	0.	751201.	0.	0.372	16.90	0.484

Alt IDEA (Sized)

CLIMB	10000.	0.547	304213.	3487.	117917.	81.	100.	11.8	693.9	0.	751201.	0.	0.315	15.30	0.529
CRUISE	30000.	0.695	300726.	0.	117917.	0.	100.	0.0	693.9	0.	-751101.	0.	0.442	18.30	0.535
DESCENT	30000.	0.750	300726.	269.	118186.	53.	153.	7.9	701.8	0.	751503.	0.	0.315	15.27	-1.053
DECEL	10000.	0.547	300457.	36.	118222.	4.	157.	0.8	702.6	0.	751503.	0.	0.373	16.90	-18.282
DESCENT	10000.	0.456	300421.	332.	118554.	30.	187.	6.6	709.2	0.	751503.	0.	0.444	18.50	6.544
CRUISE	30000.	0.690	300089.	276.	118830.	13.	200.	1.9	711.1	0.	-751101.	0.	0.447	18.39	0.533
CRUISE	1500.	0.378	299812.	4679.	123510.	0.	200.	32.0	743.1	0.	-751101.	0.	0.464	18.85	0.556

WTO = 418644.1 FUEL A=123537.1 FUEL R=123509.7

Alt IDEA (Sized)

ALTERNATE MISSION NO. 1 SUMMARY

ATX-350I, AVRG STAGE L = 2500 NMI, M = 0.80, 65% LF

SEGMENT	INIT ALTITUDE (FT)	INIT MACH NO	INIT WEIGHT (LB)	SEGMT FUEL (LB)	TOTAL FUEL (LB)	SEGMT DIST (N MI)	TOTAL DIST (N MI)	SEGMT TIME (MIN)	TOTAL TIME (MIN)	EXTERN STORE TAB ID	ENGINE THRUST TAB ID	EXTERN F TANK TAB ID	AVG CL	AVG L/D RATIO	AVG SFC (FF/T)
TAKEOFF POWER 1	0.	0.0	335569.	0.	0.	0.	0.	0.0	0.0	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	335569.	513.	513.	0.	0.	1.0	1.0	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	335056.	1561.	2075.	18.	18.	4.1	5.1	0.	751201.	0.	0.494	19.36	0.458
ACCEL	10000.	0.456	333494.	238.	2312.	4.	22.	0.7	5.7	0.	751201.	0.	0.408	17.80	0.484
CLIMB	10000.	0.547	333256.	7654.	9966.	229.	250.	31.0	36.8	0.	751201.	0.	0.409	17.26	0.538
CRUISE	40000.	0.800	325602.	0.	9966.	0.	250.	0.0	36.8	0.	-751101.	0.	0.578	19.61	0.541
ACCEL	40000.	0.800	325602.	0.	9966.	0.	250.	0.0	36.8	0.	751201.	0.	0.578	19.61	0.540
ACCEL	40000.	0.800	325602.	0.	9966.	0.	250.	0.0	36.8	0.	751201.	0.	0.578	19.61	0.540
CRUISE	40000.	0.800	325602.	38024.	47990.	2050.	2300.	268.0	304.7	0.	-751101.	0.	0.577	19.51	0.542
CLIMB	42000.	0.800	287579.	0.	47990.	0.	2300.	0.0	304.7	0.	751201.	0.	0.0	0.0	0.0
CRUISE	42000.	0.800	287579.	0.	47990.	0.	2300.	0.0	304.7	0.	-751101.	0.	0.562	19.35	0.544
CLIMB	42000.	0.800	287578.	0.	47990.	0.	2300.	0.0	304.7	0.	751201.	0.	0.0	0.0	0.0
CRUISE	42000.	0.800	287578.	0.	47990.	0.	2300.	0.0	304.7	0.	-751101.	0.	0.562	19.35	0.544
DESCENT	42000.	0.800	287578.	1219.	49210.	130.	2430.	17.8	322.5	0.	751503.	0.	0.373	16.24	-0.157
DECEL	10000.	0.547	286359.	35.	49245.	4.	2434.	0.8	323.3	0.	751503.	0.	0.356	16.41	-18.356
DESCENT	10000.	0.456	286323.	324.	49569.	29.	2463.	6.4	329.7	0.	751503.	0.	0.423	18.05	6.541
CRUISE	42000.	0.800	285999.	645.	50214.	37.	2500.	4.8	334.5	0.	-751101.	0.	0.559	19.32	0.545
LOITER	1500.	0.280	285354.	356.	50570.	0.	2500.	3.0	337.5	0.	-751101.	0.	0.811	19.79	0.494
CRUISE	1500.	0.378	284998.	288.	50858.	0.	2500.	2.0	339.5	0.	-751101.	0.	0.445	18.48	0.560
RESET	0.	0.0	284710.	0.	50858.	-2500.	0.	0.0	339.5	0.	0.	0.	0.0	0.0	0.0
CRUISE	42001.	0.800	284710.	4522.	55380.	0.	0.	34.0	373.5	0.	-751101.	0.	0.552	19.26	0.545
TAKEOFF POWER 1	0.	0.0	280188.	0.	55380.	0.	0.	0.0	373.5	0.	751701.	0.	0.0	0.0	0.545
POWER 2	0.	0.0	280188.	513.	55893.	0.	0.	1.0	374.5	0.	751401.	0.	0.0	0.0	0.303
CLIMB	0.	0.378	279675.	1242.	57136.	14.	14.	3.2	377.7	0.	751201.	0.	0.410	17.76	0.458
ACCEL	10000.	0.456	278432.	189.	57325.	3.	17.	0.5	378.3	0.	751201.	0.	0.341	15.97	0.484

Alt IDEA (Sized)

CLIMB	10000.	0.547	278244.	3261.	60586.	77.	94.	11.1	389.4	0.	751201.	0.	0.291	14.42	0.529
CRUISE	30000.	0.660	274982.	122.	60708.	6.	100.	0.9	390.3	0.	-751101.	0.	0.448	18.30	0.527
DESCENT	30000.	0.750	274860.	251.	60959.	49.	149.	7.3	397.6	0.	751503.	0.	0.288	14.29	-1.052
DECEL	10000.	0.547	274609.	34.	60994.	4.	153.	0.8	398.4	0.	751503.	0.	0.341	15.97	-18.432
DESCENT	10000.	0.456	274574.	317.	61311.	28.	182.	6.3	404.7	0.	751503.	0.	0.406	17.65	6.539
CRUISE	30000.	0.660	274257.	366.	61677.	18.	200.	2.8	407.5	0.	-751101.	0.	0.447	18.28	0.527
CRUISE	1500.	0.375	273891.	4493.	66170.	0.	200.	32.0	439.5	0.	-751101.	0.	0.431	18.17	0.563

Alt IDEA (Sized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)

	C/SM***	PERCENT
FLIGHT CREW	0.50913	15.27814
FUEL AND OIL	1.44174	43.26404
INSURANCE	0.04715	1.41477
DEPRECIATION	0.74515	22.36060
MAINTENANCE	0.58926	17.68250
TOTAL DOC	3.33242	100.000

INDIRECT OPERATIONAL COST (IOC)

	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.28371	10.39872
AIRCRAFT CONTROL	0.00854	0.31312
CABIN ATTENDANT	0.50657	18.56696
FOOD AND BEVERAGE	0.18145	6.65052
PASSENGER HANDLING	0.41303	15.13872
CARGO HANDLING	0.14903	5.46233
OTHER PASSENGER EXPENSE	0.99450	36.45099
OTHER CARGO EXPENSE	0.02268	0.83136
GENERAL + ADMINISTRATION	0.16881	6.18748
TOTAL IOC	2.72832	100.000

MISC. DATA

FLIGHT DISTANCE (N. MI.)	4600.00
BLOCK FUEL (LBS)	104103.31
BLOCK TIME (HRS)	10.59
FLIGHT TIME (HRS)	10.26
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4618.62
FLIGHTS PER A/C PER YEAR	436.11
FARE ($)	462.19
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	484.54	27.26	457.28	241.41	58.14	212.77	-219.32	6.26
2	13.0	8.0	1259.80	98.12	1161.68	627.67	141.87	553.21	-275.64	6.41
3	20.4	7.0	1974.49	209.18	1765.31	983.76	209.03	867.05	-295.24	6.61
4	23.0	0.0	2228.88	334.56	1894.32	1110.50	212.81	978.76	-134.00	6.95
5	23.0	0.0	2228.88	459.93	1768.94	1110.50	186.06	978.76	-107.26	7.45
6	23.0	0.0	2228.88	585.31	1643.57	1110.50	159.32	978.76	-80.51	8.02
7	23.0	0.0	2228.88	710.68	1518.20	1110.50	132.57	978.76	-53.76	8.68
8	23.0	0.0	2228.88	836.05	1392.82	1110.50	105.82	978.76	-39.98	8.53
9	23.0	0.0	2228.88	961.43	1267.45	1110.50	79.08	978.76	-26.60	8.32
10	23.0	0.0	2228.88	1086.80	1142.07	1110.50	52.33	978.76	-13.23	8.06
11	23.0	0.0	2228.88	1212.18	1016.70	1110.50	25.58	978.76	62.16	7.74
12	23.0	0.0	2228.88	1337.55	891.33	1110.50	8.14	978.76	132.91	7.85
13	23.0	0.0	2228.88	1462.92	765.95	1110.50	-0.00	978.76	191.25	8.60
14	23.0	0.0	2228.88	1588.30	640.58	1110.50	-0.00	978.76	191.25	10.28
15	23.0	0.0	2228.88	1713.67	515.20	1110.50	-0.00	978.76	191.25	12.79
16	23.0	0.0	2228.88	1839.05	389.83	1110.50	-0.00	978.76	191.25	16.90

AVG ROI OVER THE 16 YEAR PERIOD = 8.09 PERCENT

Alt IDEA (Sized)

OPERATIONAL COSTS

DIRECT OPERATIONAL COST (DOC)	C/SM***	PERCENT
FLIGHT CREW	0.53006	15.47397
FUEL AND OIL	1.31191	38.29837
INSURANCE	0.05472	1.59756
DEPRECIATION	0.86492	25.24953
MAINTENANCE	0.66388	19.38054
TOTAL DOC	3.42551	100.000

INDIRECT OPERATIONAL COST (IOC)	C/SM***	PERCENT
SYSTEM	0.0	0.0
LOCAL	0.52203	14.92434
AIRCRAFT CONTROL	0.01572	0.44939
CABIN ATTENDANT	0.52739	15.07770
FOOD AND BEVERAGE	0.18891	5.40071
PASSENGER HANDLING	0.75998	21.72722
CARGO HANDLING	0.27421	7.83958
OTHER PASSENGER EXPENSE	0.99450	28.43196
OTHER CARGO EXPENSE	0.02268	0.64847
GENERAL + ADMINISTRATION	0.19240	5.50070
TOTAL IOC	3.49782	100.000

MISC. DATA	
FLIGHT DISTANCE (N. MI.)	2500.00
BLOCK FUEL (LBS)	51480.47
BLOCK TIME (HRS)	5.99
FLIGHT TIME (HRS)	5.66
AVG STAGE LENGTH (N. MI.)	2500.00
AVG CARGO PER FLIGHT	17798.23
UTILIZATION (HRS PER YR)	4142.61
FLIGHTS PER A/C PER YEAR	691.32
FARE ($)	299.45
FUEL COST ($/LB)	0.22200

*** - CENTS PER SEAT N. MILE

RATE OF RETURN ON INVESTMENT

YEAR	AVG NO AIRCRAFT DURING YEAR	AIRCRAFT ADDED DURING YEAR	AVERAGE INVESTMENT DURING YEAR	CUMULATIVE DEPRECIATION	AVERAGE BOOK VALUE OF FLEET	REVENUE	INTEREST EXPENSE	OPERATING EXPENSE	CASH FLOW	ROI
			$M	$M	$M	$M	$M	$M	$M	PERCENT
1	5.0	8.0	484.54	27.26	457.28	245.93	58.14	209.40	-211.43	7.99
2	13.0	8.0	1259.80	98.12	1161.68	639.42	141.87	544.43	-255.12	8.18
3	20.4	7.0	1974.49	209.18	1765.31	1002.17	209.03	853.30	-263.07	8.43
4	23.0	0.0	2228.88	334.56	1894.32	1131.29	212.81	963.23	-97.69	8.87
5	23.0	0.0	2228.88	459.93	1768.94	1131.29	186.06	963.23	-70.94	9.50
6	23.0	0.0	2228.88	585.31	1643.57	1131.29	159.32	963.23	-48.57	9.96
7	23.0	0.0	2228.88	710.68	1518.20	1131.29	132.57	963.23	-35.19	9.90
8	23.0	0.0	2228.88	836.05	1392.82	1131.29	105.82	963.23	-21.82	9.83
9	23.0	0.0	2228.88	961.43	1267.45	1131.29	79.08	963.23	-8.45	9.75
10	23.0	0.0	2228.88	1086.80	1142.07	1131.29	52.33	963.23	4.93	9.65
11	23.0	0.0	2228.88	1212.18	1016.70	1131.29	25.58	963.23	80.32	9.52
12	23.0	0.0	2228.88	1337.55	891.33	1131.29	8.14	963.23	151.06	9.88
13	23.0	0.0	2228.88	1462.92	765.95	1131.29	-0.00	963.23	209.40	10.97
14	23.0	0.0	2228.88	1588.30	640.58	1131.29	-0.00	963.23	209.40	13.12
15	23.0	0.0	2228.88	1713.67	515.20	1131.29	-0.00	963.23	209.40	16.31
16	23.0	0.0	2228.88	1839.05	389.83	1131.29	-0.00	963.23	209.40	21.55

AVG ROI OVER THE 16 YEAR PERIOD = 10.00 PERCENT

Alt IDEA (Sized)

SUMMARY ID NO. 1 A S S E T P A R A M E T R I C A N A L Y S I S MAY 25 1984

AIRCRAFT MODEL --CL1633-11-1
I.O.C. DATE --1990
DESIGN SPEED --SUBSONIC

ENGINE I.D. -- 751000
SLS SCALE 1.0 = 39000
NUMBER OF ENGINES = 3.

WING QUARTER CHORD SWEEP = 25.00 DEG
WING TAPER RATIO = 0.246

1 W/S	131.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
2 T/W	0.249	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
3 AR	12.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
4 T/C	10.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
5 SWEEP	25.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
6 FPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
7 OPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
8 TIT	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
9 NPR	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
10 AUG T	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
11 RADIUS N. MI	4600	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
12 GROSS WEIGHT	418644	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
13 FUEL WEIGHT	123537	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
14 OP. WT. EMPTY	221607	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
15 ZERO FUEL WT.	295107	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
16 ENGINE SCALE	0.891	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
17 THRUST/ENGINE	34747	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
18 WING AREA	3196.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.	0.
19 WING SPAN	195.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
20 H. TAIL AREA	460.7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
21 V. TAIL AREA	469.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
22 ENG. LENGTH	14.54	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
23 ENG. DIAMETER	6.51	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
24 BODY LENGTH	201.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
25 WING FUEL LIMIT	1.055	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
COST DATA																
26 RDTE - BIL.	3.042	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
27 FLYAWAY - MIL.	82.76	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
28 INVESTMNT-BIL.	29.072	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
29 DOC - C/SM	3.332	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
30 IOC - C/SM	2.728	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
31 ROI A.T. - O/O	8.09	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
MISSION PARAMETERS																
32 MISN V1(1,1)	36000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
33 MISN V2(1,1)	104103	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
34 MISN V1(2,1)	40000	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
35 MISN V2(2,1)	51480	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
CONSTRAINT OUTPUT																
36 TAKEOFF DST(1)	9181	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
37 CLIMB GRAD(1)	0.1099	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
38 TAKEOFF DST(2)	9303	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
39 CLIMB GRAD(2)	0.0448	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
40 CTOL LNDG D(1)	6119	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
41 AP SPEED-KT(1)	135.3	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
42 CTOL LNDG D(2)	6800	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
43 AP SPEED-KT(2)	145.6	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
44 CTOL LNDG D(3)	7480	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
45 AP SPEED-KT(3)	155.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
46 SEP(1) - FPS	1	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
47 SEP(2) - FPS	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
48 SEP(3) - FPS	4	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
49 SEP(4) - FPS	2	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0

Alt IDEA (Sized)

APPENDIX B

SYSTEM RELIABILITY ASSESSMENT
SELECTED ELECTRIC POWER GENERATION AND DISTRIBUTION SYSTEM
INTEGRATED DIGITAL ELECTRIC AIRCRAFT

By
William Ng

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Introduction

The purpose of this analysis is to ascertain if the selected Integrated Digital Electric Aircraft (IDEA) power generation system and distribution system is certifiable under the FAA guidelines. The certifiability criterion is that the probability of losing flight critical power shall be less than 10^{-9} on a per flight hour basis.

The selected power generation system is a partial converted variable voltage and variable frequency system (VVVF) and the selected power distribution is a ring main arrangement, see Figure 1. This combined arrangement enables the power for the flight control actuators to be tapped off directly from the ring/bus, see Figure 1. The power for flight control computers are tapped off the constant voltage constant frequency (CVCF) buses which are rectified and inverted from the VVVF buses, see Figure 2. Thus power off the left and right ring, VVVF buses and the CVCF buses are considered flight critical electric power.

Thus, this analysis will assess:

(1) The probability of losing power to each of the VVVF buses.

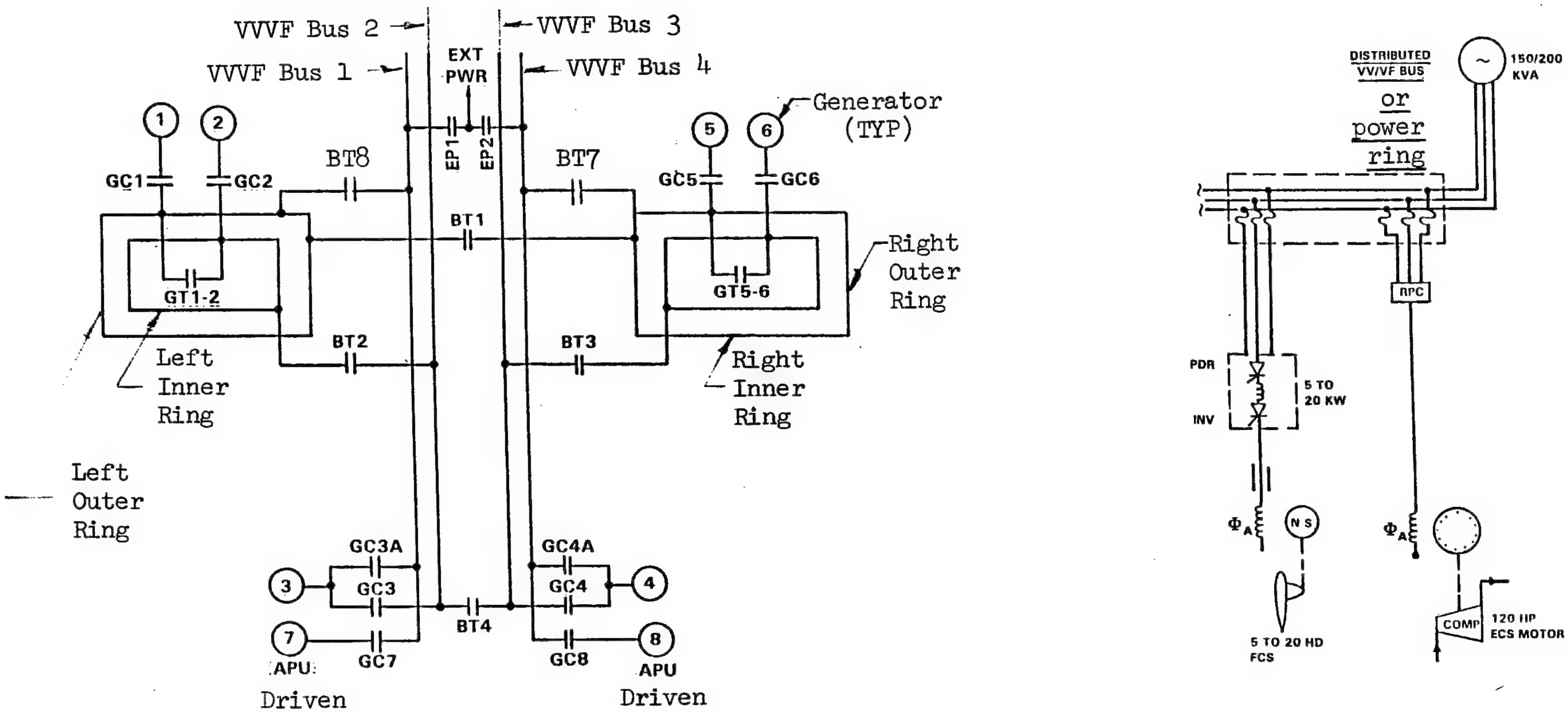
(2) The probability of losing power to left outer ring, left inner ring, right outer ring and right inner ring.

(3) The probability of losing power to flight control computer (FCS) 1, 2, 3 and 4.

(4) The probability of losing all electric power.

Result

The intent of the FAA guidelines has been satisfied. All the events which have been assessed fall into the extremely improbable category. The probability of the occurrence of the events on a per flight hour basis are as follows.



Power Distribution System

Power Utilization

Figure 1. IDEA Selected Electric Power Generation and Distribution System

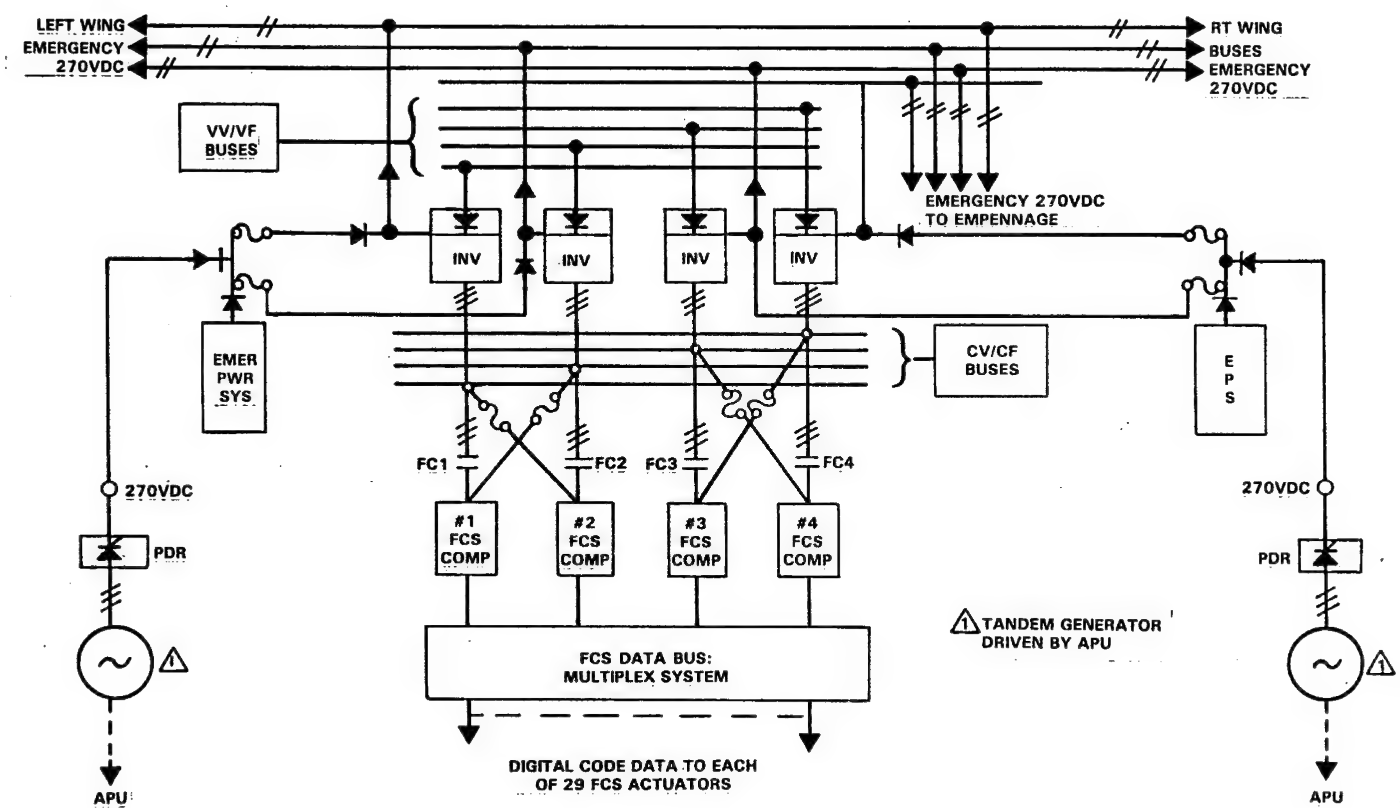


Figure 2. IDEA Flight Critical Electric Power

The probability of losing power to a VVVF Bus

VVVF Bus 1 = 3.50×10^{-13}

VVVF Bus 2 = 2.04×10^{-15}

VVVF Bus 3 = 2.04×10^{-5}

VVVF Bus 4 = 3.50×10^{-13}

The probability of losing power to

Left Outer Ring = 1.36×10^{-15}

Left Inner Ring = 2.60×10^{-15}

Right Outer Ring = $1.36 \times 10\text{-}^{15}$

Right Inner Ring = 2.60×10^{-15}

The probability of losing power to flight control computer

Computer 1 = 1.36×10^{-9}

Computer 2 = 1.36×10^{-9}

Computer 3 = 1.36×10^{-9}

Computer 4 = 1.36×10^{-9}

The probability of losing all electric power is 1.05×10^{-15}.

Discussion and Analysis

The final configuration for the power generation system and distribution system is shown in Figure 1 and Figure 2.

The system is symmetrical; the result of the analysis done on the left side is also applicable to the right side. Thus, only the left hand rings and VV/VF Bus 1 and Bus 2 will be analyzed.

The assumptions required for this analysis are:

1) All failures are primary failures

2) The digital power/load management system is perfect

3) No generator is switched on to a bus or ring parallel with another generator

4) Any single generator is sufficient to meet the aircraft load demand

5) The systems are operative before take-off.

6) All failures are exponentially distributed.

7) Contacts EP1 and EP2 are for external power connection only.

The technique used for this assessment is fault tree analysis. The system configuration is translated into a logic tree. Each undesirable event, such as no power to VVVF Bus, no power to FCS 1 or no power to power ring, is treated as the top event for the logic tree.

There are three outputs associated with each logic tree analysis, (1) the logic structure, (2) the failure probability time history, and (3) the cut sets.

The logic structure documents the configuration of the system under evaluation.

The failure probability time history provides a quantitative measure which could be viewed as figure of merit or as a relative comparison between alternate configurations.

The cut sets are failure paths of the system's logic structure. A cut set is a collection of basic events (component failures) whose presence will cause the top event to occur. The cut sets serve as a qualitative measure. One could interpret that cut sets with one element are

"single point failures" of the system, cut set with two elements are "double point failures", etc. Only the first twenty-five top ranked cut sets are shown for each logic tree in this analysis.

VVVF Bus 1

The associated outputs are Table 1, Figure 3 and Table 2. The probability of losing power to VVVF Bus 1 is 3.5×10^{-13}, on a per flight hour basis. This result is also applicable to VVVF Bus 4.

VVVF Bus 2

The associated outputs are Table 3, Figure 4 and Table 4. The probability of losing power to VVVF Bus 2 is 2.04×10^{-15}, on a per flight hour basis. This result is also applicable to VVVF Bus 3.

Left Outer Ring

The associated outputs are Table 5, Figure 5 and Table 6. The probability of losing power to the left outer ring is 1.36×10^{-15}, on a per flight hour basis. This result is also applicable to right outer ring.

Left Inner Ring

The associated outputs are Table 7, Figure 6 and Table 8. The probability of losing power to the inner ring is 2.60×10^{-15} on a per flight hour basis. This result is also applicable to right inner ring.

No Power to FCS 1

The associated outputs are Table 9, Figure 7 and Table 10. The probability of losing power to FCS 1 is 1.36×10^{-9}, on a per flight hour basis. This result is also applicable to FCS 4.

No Power to FCS 2

The associated outputs are Table 11, Figure 8 and Table 12. The probability of losing power to FCS 2 is 1.36×10^{-9} on a per flight hour basis. This result is also applicable to FCS 3.

No single point failure which could cause the occurrence of the top event was found in the previous analysis.

The cut set, all engines out and the APU is not functioning, will cause the event, loss of all electric power to occur. The probability of occurrence of this event is 1.05×10^{-15}, on a per flight hour basis.

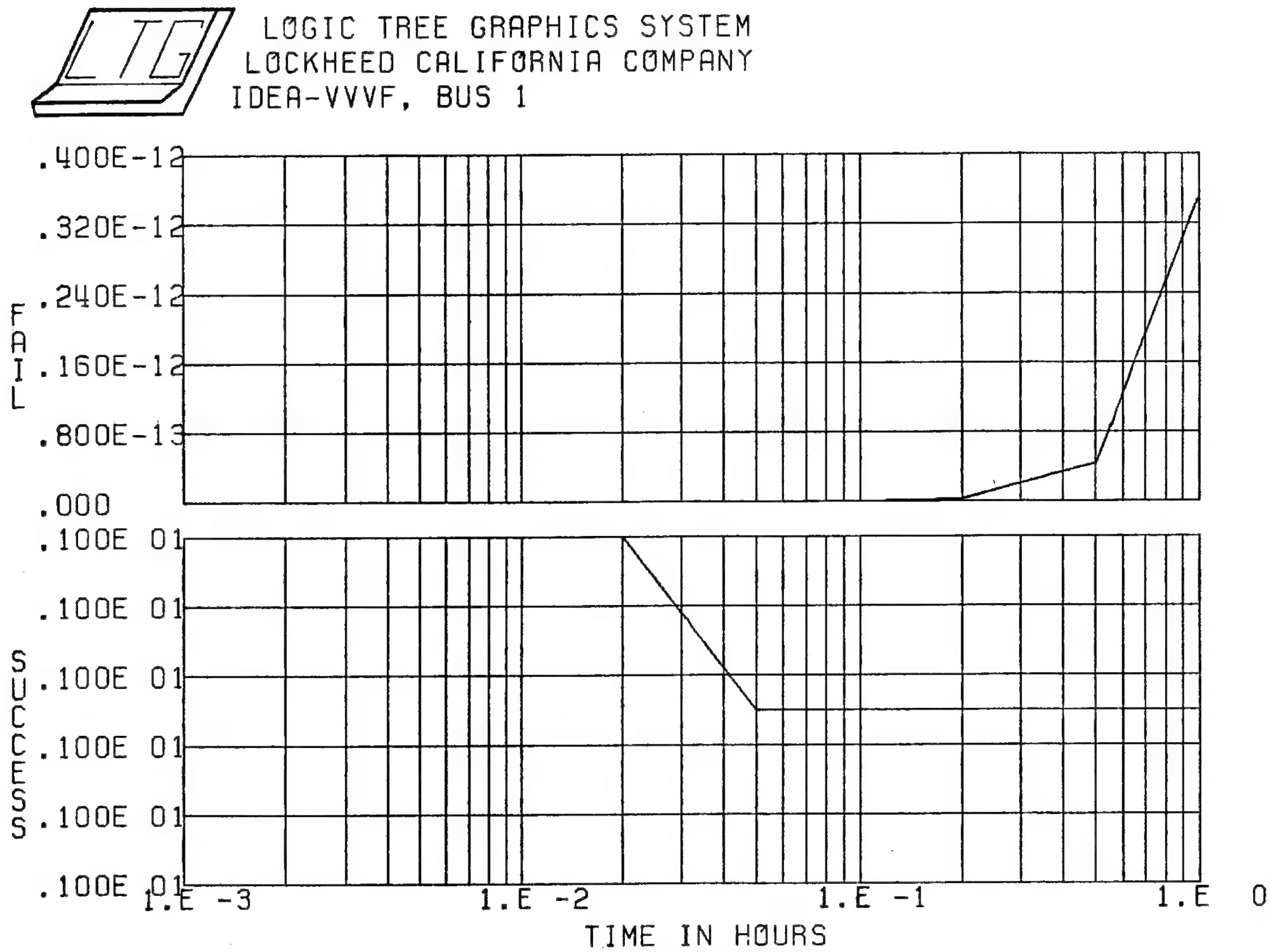
The emergency power system (EPS) as the power source for the all engines and APU out condition was not included in the assessment. The FAA guideline is met without EPS. It would be a plus if the EPS is included as a power source.

Failure rates used in this assessment are either point estimates derived from the L1011 experience or quoted by the manufacturer. The reference for these failure rates are shown in Table 13.

Table 1. Logic Structure, VVVF Bus 1

```
*********************************************************************
******* CUTSETS OBTAINED BY MOCUS - IDEA-VVVF, BUS 1 4/23/84 *******
*********************************************************************
  99 ELEMENTS USED TO DESCRIBE THIS TREE -    39 GATES ARE DEFINED
```

GATE	TYPE	# GATES	# COMP.	INPUTS			
T1	OR	1	1	INV1	1		
1	AND	3	0	A1	A2	A3	
A1	OR	0	3	GC3A	GEN3	ENG2	
A2	OR	1	1	BT8	A4		
A3	OR	0	3	GC7	APU	APUC	
A4	AND	3	0	A5	A6	A8	
A5	OR	1	1	GC12	N1		
A6	OR	0	3	GC1	GEN1	ENG1	
A8	OR	1	1	BT1	A11		
A11	AND	3	0	N11	A12	A13	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A12	OR	1	1	GT56	A14		
A13	OR	0	3	GC5	ENG3	GEN5	
A14	AND	2	0	A16	A17		
A16	OR	1	1	BT3	A21		
A17	OR	0	3	GC6	ENG3	GEN6	
N1	AND	2	0	A9	A7		
A9	OR	0	3	GC2	ENG1	GEN2	
A7	OR	1	1	BT2	N2		
N2	AND	2	0	A10	N3		
A10	OR	0	3	C3A	GEN3	ENG2	
N3	AND	2	0	N4	N5		
N4	OR	1	1	BT3	N6		
N5	OR	0	3	GC4	ENG2	GEN4	
N6	AND	2	0	N7	N8		
N7	OR	0	3	GC6	ENG3	GEN6	
N8	OR	1	1	GT56	N9		
N9	AND	2	0	N10	N11		
N10	OR	0	3	GC5	ENG3	GEN5	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A21	OR	2	0	N15	N16		
N15	OR	0	3	GC4	GEN4	ENG2	
N16	OR	0	4	BT4	GC3	ENG2	GEN3



TIME IN HOURS	FAIL PROB.	SUCCESS PROB.
0.0	0.0	1.00000000E 00
9.99999931E-04	0.0	1.00000000E 00
1.99999986E-03	0.0	1.00000000E 00
4.99999896E-03	0.0	1.00000000E 00
9.99999791E-03	0.0	1.00000000E 00
1.99999996E-02	0.0	1.00000000E 00
4.99999970E-02	2.77555756E-17	9.99999940E-01
9.99999642E-02	3.19189120E-16	9.99999940E-01
1.99999988E-01	2.73392420E-15	9.99999940E-01
5.00000000E-01	4.36595204E-14	9.99999940E-01
1.00000000E 00	3.50372509E-13	9.99999940E-01

Figure 3. Failure Probability - Time History VVVF Bus 1

Table 2. Cut Set, VVVF Bus 1

IDEA-VVVF, BUS 1 4/23/84 APRIL 24, 1984

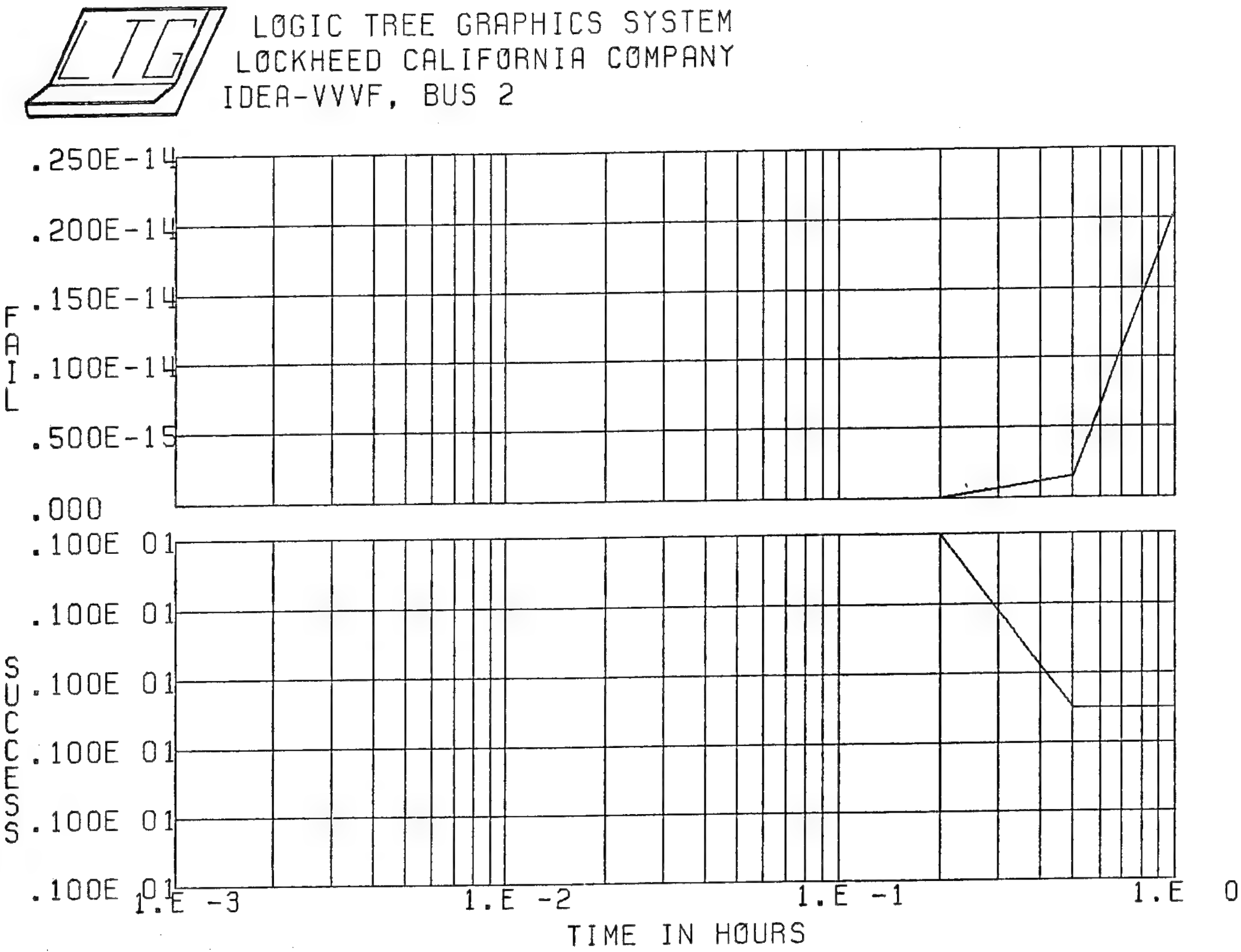
MINIMAL CUT SET DATA IN DESCENDING ORDER OF PROBABILITTY

NO.	CUT SET NO.	MAX FAILURE PROB.	COMPONENTS CONTAINED IN SET				
1	4	.14160428E-12	ENG2	BT8	APU		
2	6	.12090941E-12	GEN3	BT8	APU		
3	1	.44258644E-13	ENG2	BT8	APUC		
4	2	.37790429E-13	GEN3	BT8	APUC		
5	8	.24727858E-14	GC3A	BT8	APU		
6	10	.10540982E-14	ENG2	APU	ENG1	ENG3	
7	3	.77287310E-15	GC3A	BT8	APUC		
8	5	.66983005E-15	ENG2	BT8	GC7		
9	7	.57193719E-15	GEN3	BT8	GC7		
10	11	.32946003E-15	ENG2	APUC	ENG1	ENG3	
11	9	.11697006E-16	GC3A	BT8	GC7		
12	73	.99899481E-19	GEN3	APU	ENG1	GEN4	ENG3
13	55	.99899481E-19	ENG2	APU	GEN2	GEN1	ENG3
14	64	.31223751E-19	ENG2	APUC	GEN2	GEN1	ENG3
15	60	.31223751E-19	GEN3	APUC	ENG1	GEN4	ENG3
16	76	.20431006E-20	GEN3	APU	ENG1	ENG3	BT2
17	65	.20431006E-20	ENG2	APU	GC2	GEN1	ENG3
18	58	.20431006E-20	ENG2	APU	GEN2	GC1	ENG3
19	56	.20431006E-20	GEN3	APU	ENG1	ENG3	GC4
20	54	.20431006E-20	ENG2	APU	GC12	GEN1	ENG3
21	50	.20431006E-20	GEN3	APU	GC12	ENG1	ENG3
22	75	.17445102E-20	GEN3	APU	GC12	GEN1	ENG3
23	74	.63857435E-21	ENG2	APUC	GEN2	GC1	ENG3
24	72	.63857435E-21	ENG2	APUC	GC12	GEN1	ENG3
25	71	.63857435E-21	GEN3	APUC	ENG1	ENG3	BT2

Table 3. Logic Structure, VVVF Bus 2

```
*********************************************************************
******* CUTSETS OBTAINED BY MOCUS - IDEA-VVVF, BUS 2          *******
*********************************************************************
  97 ELEMENTS USED TO DESCRIBE THIS TREE -   38 GATES ARE DEFINED
```

GATE	TYPE	# GATES	# COMP.	INPUTS			
2	AND	3	0	B1	B2	N3	
B1	OR	0	3	GC3A	GEN3	ENG2	
B2	OR	1	1	BT2	B4		
B4	AND	2	0	B5	B6		
B5	OR	0	3	GC2	ENG1	GEN2	
B6	OR	1	1	GC1	B7		
N3	AND	2	0	N4	N5		
N4	OR	1	1	BT3	N6		
N5	OR	0	3	GC4	ENG2	GEN4	
N6	AND	2	0	N7	N8		
N7	OR	0	3	GC6	ENG3	GEN6	
N8	OR	1	1	GT56	N9		
N9	AND	2	0	N10	N11		
N10	OR	0	3	GC5	ENG3	GEN5	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
B7	AND	3	0	B8	B16	A8	
B8	OR	0	3	GC1	ENG1	GEN1	
B16	OR	1	1	BT8	B12		
A8	OR	1	1	BT1	A11		
A11	AND	3	0	N11	A12	A13	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A12	OR	1	1	GT56	A14		
A13	OR	0	3	GC5	ENG3	GEN5	
A14	AND	2	0	A16	A17		
A16	OR	1	1	BT3	A21		
A17	OR	0	3	GC6	ENG3	GEN6	
A21	OR	2	0	N15	N16		
N15	OR	0	3	GC4	GEN4	ENG2	
N16	OR	0	4	BT4	GC3	ENG2	GEN3
B12	AND	2	0	B21	B22		
B21	OR	0	3	GC7	APUC	APU	
B22	OR	0	3	GC3A	ENG2	GEN3	



TIME IN HOURS	FAIL PROB.	SUCCESS PROB.
0.0	0.0	1.00000000E 00
9.99999931E-04	0.0	1.00000000E 00
1.99999986E-03	0.0	1.00000000E 00
4.99999896E-03	0.0	1.00000000E 00
9.99999791E-03	0.0	1.00000000E 00
1.99999996E-02	0.0	1.00000000E 00
4.99999970E-02	0.0	1.00000000E 00
9.99999642E-02	0.0	1.00000000E 00
1.99999988E-01	0.0	1.00000000E 00
5.00000000E-01	1.52655666E-16	9.99999940E-01
1.00000000E 00	2.04003481E-15	9.99999940E-01

Figure 4. Failure Probability - Time History VVVF Bus 2

Table 4. Cut Set VVVF Bus 2

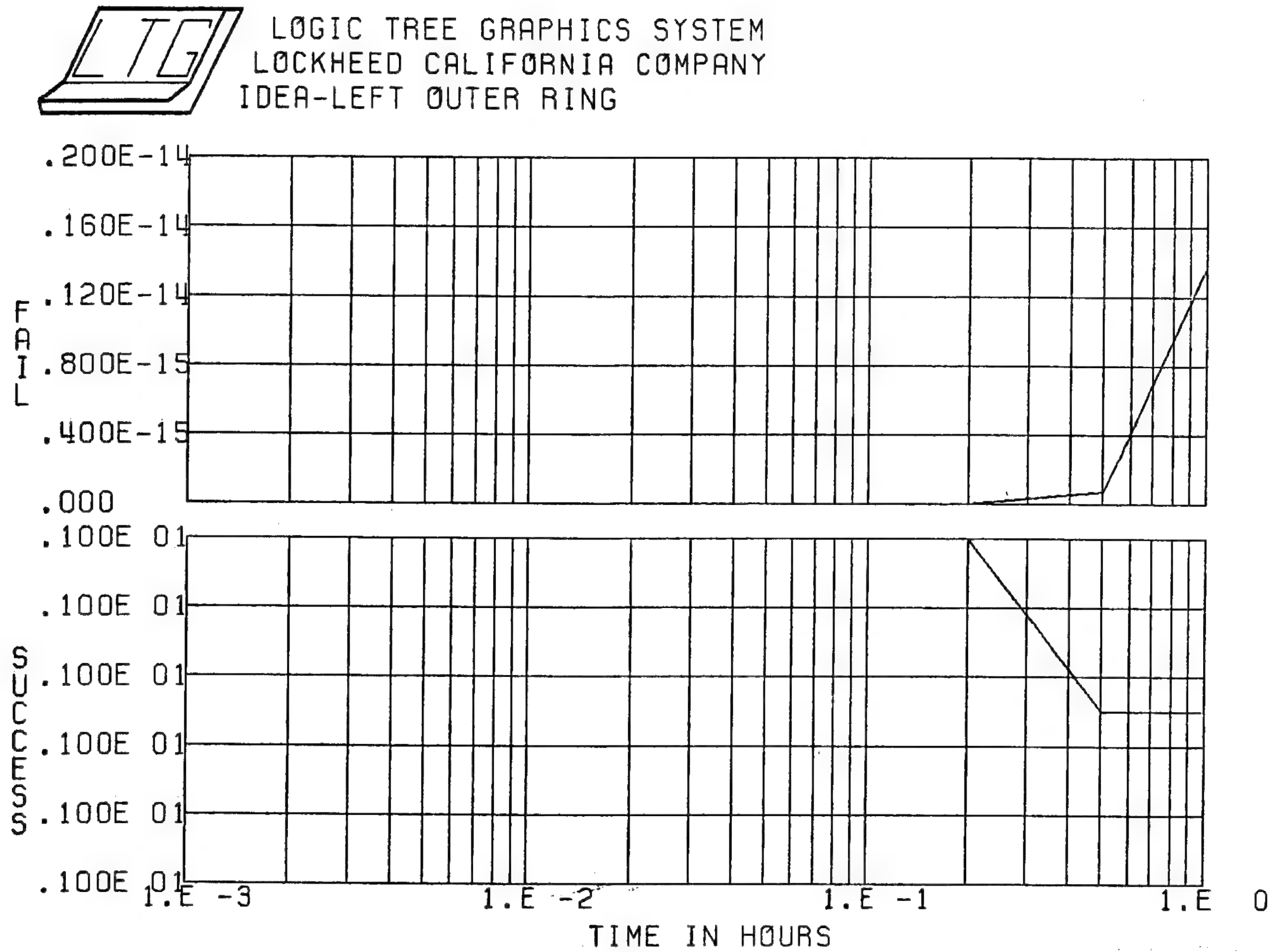
IDEA-VVVF, BUS 2 APRIL 24, 1984
MINIMAL CUT SET DATA IN DESCENDING ORDER OF PROBABILITTY

NO.	CUT SET NO.	MAX FAILURE PROB.	COMPONENTS CONTAINED IN SET				
1	14	.10540982E-14	ENG2	ENG1	ENG3	APU	
2	1	.66983005E-15	ENG2	BT2	BT3		
3	17	.32946003E-15	ENG2	ENG1	ENG3	APUC	
4	13	.18407336E-16	ENG2	BT2	ENG3	APU	
5	15	.57532455E-17	ENG2	BT2	ENG3	APUC	
6	73	.99899481E-19	ENG2	GEN2	ENG3	GEN1	APU
7	64	.99899481E-19	GEN3	ENG1	GEN4	ENG3	APU
8	16	.87072185E-19	ENG2	BT2	ENG3	GC8	
9	8	.87072185E-19	ENG2	BT3	ENG1	GC1	
10	12	.87072185E-19	ENG2	BT2	ENG3	BT7	
11	10	.87072185E-19	ENG2	BT2	GT56	ENG3	
12	5	.74346942E-19	ENG2	BT2	GT56	GEN6	
13	4	.74346942E-19	ENG2	BT3	GEN2	GC1	
14	2	.63481459E-19	GEN3	BT2	GEN4	BT3	
15	97	.31223751E-19	GEN3	ENG1	GEN4	ENG3	APUC
16	94	.31223751E-19	ENG2	GEN2	ENG3	GEN1	APUC
17	95	.20431006E-20	ENG2	GC2	ENG3	GEN1	APU
18	76	.20431006E-20	GC3A	ENG1	GEN4	ENG3	APU
19	75	.20431006E-20	GEN3	ENG1	GC4	ENG3	APU
20	72	.20431006E-20	ENG2	GEN2	ENG3	GC1	APU
21	67	.17445102E-20	GEN3	BT2	GEN4	ENG3	APU
22	11	.15205108E-20	ENG2	BT2	GT56	GC6	
23	9	.15205108E-20	ENG2	BT3	GC2	GC1	
24	3	.12982947E-20	GC3A	BT2	GEN4	BT3	
25	6	.12982947E-20	GEN3	BT2	GC4	BT3	

Table 5. Logic Structure, Left Outer Ring

```
*****************************************************************
******* CUTSETS OBTAINED BY MOCUS - IDEA-LEFT OUTER RING   *******
*****************************************************************
  97 ELEMENTS USED TO DESCRIBE THIS TREE -   38 GATES ARE DEFINED
```

GATE	TYPE	# GATES	# COMP.	INPUTS			
LOR	AND	4	0	A1	A2	A5	A8
A1	OR	0	3	GC1	GEN1	ENG1	
A2	OR	1	1	BT8	A4		
A4	AND	2	0	A6	N31		
A5	OR	1	1	GC12	N1		
A6	OR	0	3	GC3A	GEN3	ENG2	
A8	OR	1	1	BT1	A11		
A11	AND	3	0	N11	A12	A13	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A12	OR	1	1	GT56	A14		
A13	OR	0	3	GC5	ENG3	GEN5	
A14	AND	2	0	A16	A17		
A16	OR	1	1	BT3	A21		
A17	OR	0	3	GC6	ENG3	GEN6	
N1	AND	2	0	A9	A7		
A9	OR	0	3	GC2	ENG1	GEN2	
A7	OR	1	1	BT2	N2		
N2	AND	2	0	A10	N3		
A10	OR	0	3	C3A	GEN3	ENG2	
N3	AND	2	0	N4	N5		
N4	OR	1	1	BT3	N6		
N5	OR	0	3	GC4	ENG2	GEN4	
N6	AND	2	0	N7	N8		
N7	OR	0	3	GC6	ENG3	GEN6	
N8	OR	1	1	GT56	N9		
N9	AND	2	0	N10	N11		
N10	OR	0	3	GC5	ENG3	GEN5	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A21	OR	2	0	N15	N16		
N15	OR	0	3	GC4	GEN4	ENG2	
N16	OR	0	4	BT4	GC3	ENG2	GEN3
N31	OR	0	3	GC7	APU	APUC	



TIME IN HOURS	FAIL PROB.	SUCCESS PROB.
0.0	0.0	1.00000000E 00
9.99999931E-04	0.0	1.00000000E 00
1.99999986E-03	0.0	1.00000000E 00
4.99999896E-03	0.0	1.00000000E 00
9.99999791E-03	0.0	1.00000000E 00
1.99999996E-02	0.0	1.00000000E 00
4.99999970E-02	0.0	1.00000000E 00
9.99999642E-02	0.0	1.00000000E 00
1.99999988E-01	0.0	1.00000000E 00
5.00000000E-01	6.93889390E-17	9.99999940E-01
1.00000000E 00	1.36002321E-15	9.99999940E-01

Figure 5. Failure Probability - Time History Left Outer Ring

Table 6. Cut Set Left Outer Ring

IDEA-LEFT OUTER RING APRIL 25, 1984

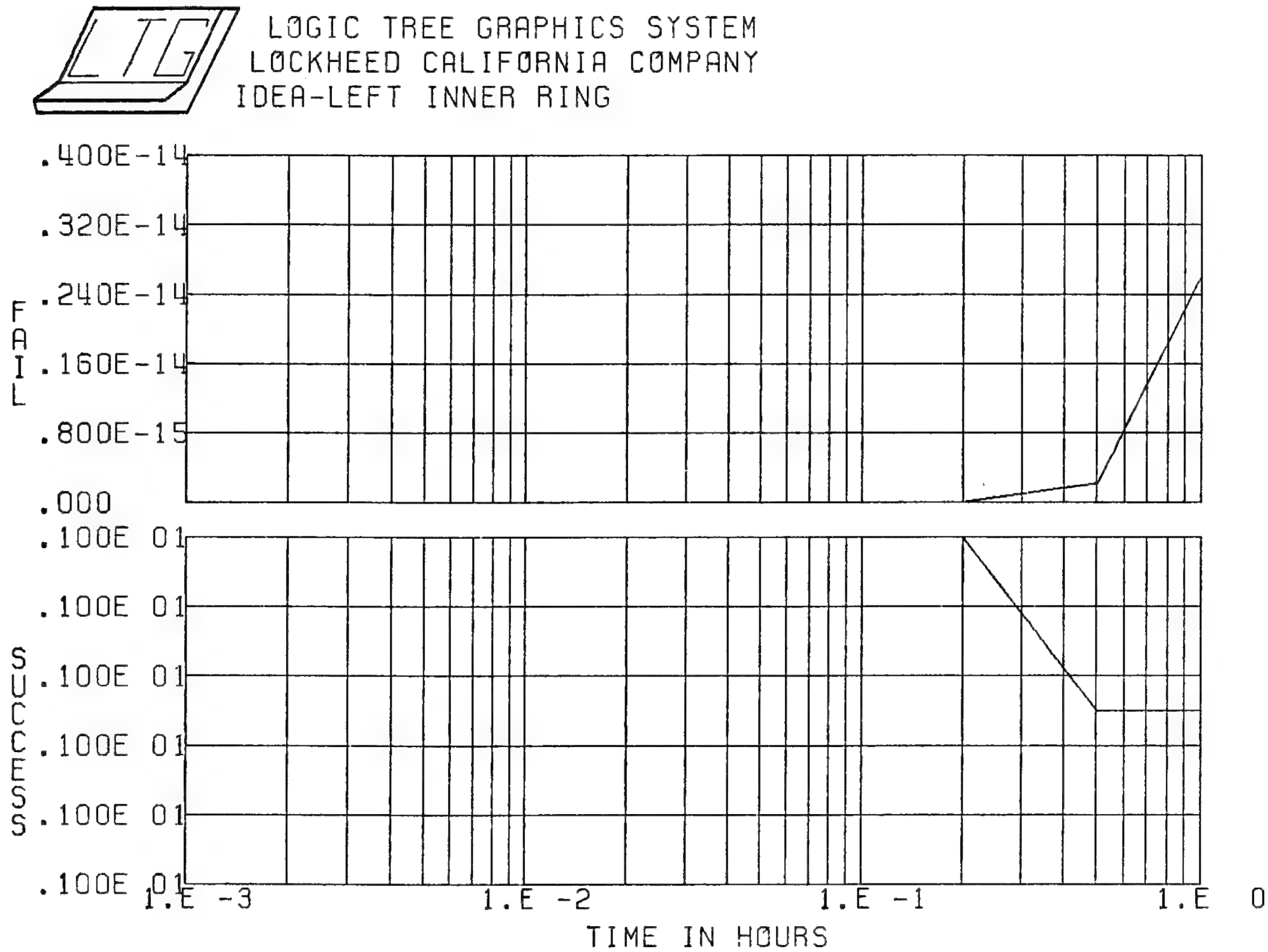
MINIMAL CUT SET DATA IN DESCENDING ORDER OF PROBABILITTY

NO.	CUT SET NO.	MAX FAILURE PROB.	COMPONENTS CONTAINED IN SET				
1	6	.10540982E-14	ENG1	ENG2	ENG3	APU	
2	5	.32946003E-15	ENG1	ENG2	ENG3	APUC	
3	69	.99899481E-19	GEN1	ENG2	GEN2	APU	ENG3
4	45	.99899481E-19	ENG1	GEN3	ENG3	APU	GEN4
5	76	.31223751E-19	ENG1	GEN3	ENG3	APUC	GEN4
6	73	.31223751E-19	GEN1	ENG2	GEN2	APUC	ENG3
7	72	.20431006E-20	GEN1	ENG2	GC12	APU	ENG3
8	71	.20431006E-20	GEN1	ENG2	GC2	APU	ENG3
9	67	.20431006E-20	ENG1	GEN3	GC12	APU	ENG3
10	64	.20431006E-20	ENG1	GEN3	ENG3	APU	BT2
11	58	.20431006E-20	GC1	ENG2	GEN2	APU	ENG3
12	57	.20431006E-20	ENG1	GEN3	ENG3	APU	GC4
13	53	.17445102E-20	GEN1	GEN3	GC12	APU	ENG3
14	1	.15205108E-20	ENG1	BT8	GC12	BT1	
15	4	.15205108E-20	ENG1	BT8	BT1	BT2	
16	2	.12982947E-20	GEN1	BT8	GC12	BT1	
17	77	.63857435E-21	ENG1	GEN3	GC12	APUC	ENG3
18	74	.63857435E-21	GEN1	ENG2	GC12	APUC	ENG3
19	63	.63857435E-21	ENG1	GEN3	ENG3	APUC	BT2
20	62	.63857435E-21	ENG1	GEN3	ENG3	APUC	GC4
21	56	.63857435E-21	GC1	ENG2	GEN2	APUC	ENG3
22	52	.63857435E-21	GEN1	ENG2	GC2	APUC	ENG3
23	70	.54524935E-21	GEN1	GEN3	GC12	APUC	ENG3
24	60	.41784586E-22	GC1	ENG2	GC2	APU	ENG3
25	54	.41784586E-22	GC1	ENG2	GC12	APU	ENG3

Table 7. Logic Structure, Left Inner Ring

```
*********************************************************************
******* CUTSETS OBTAINED BY MOCUS - IDEA-LEFT INNER RING     *******
*********************************************************************
  97 ELEMENTS USED TO DESCRIBE THIS TREE -    38 GATES ARE DEFINED
```

GATE	TYPE	# GATES	# COMP.	INPUTS			
LIR	AND	3	0	A1	A7	L1	
A1	OR	0	3	GC2	GEN2	ENG1	
A7	OR	1	1	BT2	N2		
N2	AND	2	0	A10	N3		
A10	OR	0	3	C3A	GEN3	ENG2	
N3	AND	2	0	N4	N5		
N4	OR	1	1	BT3	N6		
N5	OR	0	3	GC4	ENG2	GEN4	
N6	AND	2	0	N7	N8		
N7	OR	0	3	GC6	ENG3	GEN6	
N8	OR	1	1	GT56	N9		
N9	AND	2	0	N10	N11		
N10	OR	0	3	GC5	ENG3	GEN5	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
L1	OR	1	1	GT12	L2		
L2	AND	3	0	L3	L4	A8	
L3	OR	1	1	BT8	L5		
L4	OR	0	3	GC1	ENG1	GEN1	
A8	OR	1	1	BT1	A11		
A11	AND	3	0	N11	A12	A13	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A12	OR	1	1	GT56	A14		
A13	OR	0	3	GC5	ENG3	GEN5	
A14	AND	2	0	A16	A17		
A16	OR	1	1	BT3	A21		
A17	OR	0	3	GC6	ENG3	GEN6	
A21	OR	2	0	N15	N16		
N15	OR	0	3	GC4	GEN4	ENG2	
N16	OR	0	4	BT4	GC3	ENG2	GEN3
L5	AND	2	0	L6	L7		
L6	OR	0	3	GC3A	GEN3	ENG2	
L7	OR	0	3	GC7	APUC	APU	



TIME IN HOURS	FAIL PROB.	SUCCESS PROB.
0.0	0.0	1.00000000E 00
9.99999931E-04	0.0	1.00000000E 00
1.99999986E-03	0.0	1.00000000E 00
4.99999896E-03	0.0	1.00000000E 00
9.99999791E-03	0.0	1.00000000E 00
1.99999996E-02	0.0	1.00000000E 00
4.99999970E-02	0.0	1.00000000E 00
9.99999642E-02	0.0	1.00000000E 00
1.99999988E-01	0.0	1.00000000E 00
5.00000000E-01	2.22044605E-16	9.99999940E-01
1.00000000E 00	2.59514632E-15	9.99999940E-01

Figure 6. Failure Probability - Time History Left Inner Ring

Table 8. Cut Set Left Inner Ring

IDEA-LEFT INNER RING APRIL 25, 1984

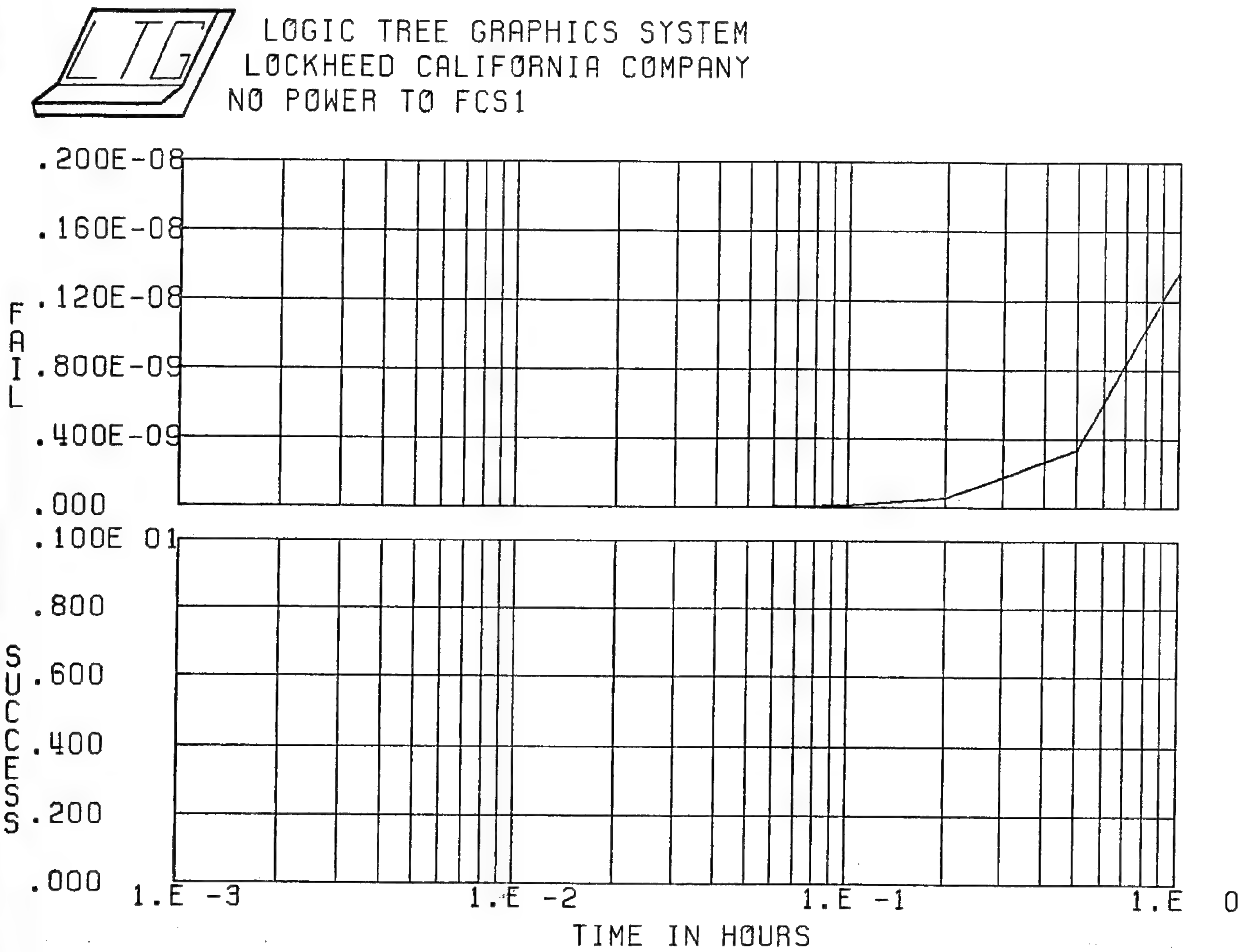
MINIMAL CUT SET DATA IN DESCENDING ORDER OF PROBABILITTY

NO.	CUT SET NO.	MAX FAILURE PROB.	COMPONENTS CONTAINED IN SET				
1	8	.10540982E-14	ENG1	ENG3	ENG2	APU	
2	1	.66983005E-15	ENG1	BT2	GT12		
3	2	.57193719E-15	GEN2	BT2	GT12		
4	9	.32946003E-15	ENG1	ENG3	ENG2	APUC	
5	3	.11697006E-16	GC2	BT2	GT12		
6	55	.99899481E-19	GEN2	ENG3	ENG2	APU	GEN1
7	4	.87072185E-19	ENG1	GT12	ENG2	BT3	
8	6	.74346942E-19	GEN2	GT12	ENG2	BT3	
9	69	.31223751E-19	GEN2	ENG3	ENG2	APUC	GEN1
10	68	.20431006E-20	ENG1	BT2	GEN3	ENG3	APU
11	62	.20431006E-20	GEN2	GT12	ENG2	ENG3	APU
12	40	.20431006E-20	GC2	ENG3	ENG2	APU	GEN1
13	34	.20431006E-20	GEN2	ENG3	ENG2	APU	GC1
14	7	.15205108E-20	GC2	GT12	ENG2	BT3	
15	5	.15205108E-20	ENG1	BT2	BT1	BT8	
16	70	.63857435E-21	GEN2	ENG3	ENG2	APUC	GC1
17	65	.63857435E-21	GEN2	GT12	ENG2	ENG3	APUC
18	50	.63857435E-21	ENG1	BT2	GEN3	ENG3	APUC
19	41	.63857435E-21	GC2	ENG3	ENG2	APUC	GEN1
20	16	.41784586E-22	ENG1	BT2	BT1	APU	ENG2
21	64	.41784586E-22	GC2	GT12	ENG2	ENG3	APU
22	59	.41784586E-22	GC2	ENG3	ENG2	APU	GC1
23	25	.41784586E-22	ENG1	BT3	ENG2	APU	BT1
24	32	.35677945E-22	ENG1	BT2	BT1	APU	GEN3
25	71	.13059839E-22	GC2	ENG3	ENG2	APUC	GC1

Table 9. Logic Structure, No Power to FCS 1

```
*********************************************************************
******* CUTSETS OBTAINED BY MOCUS - NO POWER TO FCS1 4/23/84 *******
*********************************************************************
 101 ELEMENTS USED TO DESCRIBE THIS TREE -   40 GATES ARE DEFINED
```

GATE	TYPE	# GATES	# COMP.	INPUTS			
TOP	OR	2	0	2	1		
2	AND	0	2	INV2	INV1		
1	AND	3	0	A3	A2	A1	
A3	OR	0	3	GC7	APU	APUC	
A2	OR	1	1	BT8	A4		
A4	AND	3	0	A5	A6	A8	
A5	OR	1	1	GC12	N1		
A6	OR	0	3	GC1	GEN1	ENG1	
A8	OR	1	1	BT1	A11		
A11	AND	3	0	N11	A12	A13	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A12	OR	1	1	GT56	A14		
A13	OR	0	3	GC5	ENG3	GEN5	
A14	AND	2	0	A16	A17		
A16	OR	1	1	BT3	A21		
A17	OR	0	3	GC6	ENG3	GEN6	
N1	AND	2	0	A9	A7		
A9	OR	0	3	GC2	ENG1	GEN2	
A7	OR	1	1	BT2	N2		
N2	AND	2	0	A10	N3		
A10	OR	0	3	C3A	GEN3	ENG2	
N3	AND	2	0	N4	N5		
N4	OR	1	1	BT3	N6		
N5	OR	0	3	GC4	ENG2	GEN4	
N6	AND	2	0	N7	N8		
N7	OR	0	3	GC6	ENG3	GEN6	
N8	OR	1	1	GT56	N9		
N9	AND	2	0	N10	N11		
N10	OR	0	3	GC5	ENG3	GEN5	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A21	OR	2	0	N15	N16		
N15	OR	0	3	GC4	GEN4	ENG2	
N16	OR	0	4	BT4	GC3	ENG2	GEN3
A1	OR	0	3	GC3A	GEN3	ENG2	



TIME IN HOURS	FAIL PROB.	SUCCESS PROB.
0.0	0.0	1.00000000E 00
9.99999931E-04	1.36002321E-15	9.99999940E-01
1.99999986E-03	5.44009282E-15	9.99999940E-01
4.99999896E-03	3.40005801E-14	9.99999940E-01
9.99999791E-03	1.36002321E-13	9.99999940E-01
1.99999996E-02	5.44050915E-13	9.99999940E-01
4.99999970E-02	3.40033557E-12	9.99999940E-01
9.99999642E-02	1.36015643E-11	9.99999940E-01
1.99999988E-01	5.44075895E-11	9.99999940E-01
5.00000000E-01	3.40070194E-10	9.99999940E-01
1.00000000E 00	1.36043177E-09	9.99999940E-01

Figure 7. Failure Probability - Time History No Power to FCS 1

Table 10. Cut Set, No Power to FCS 1

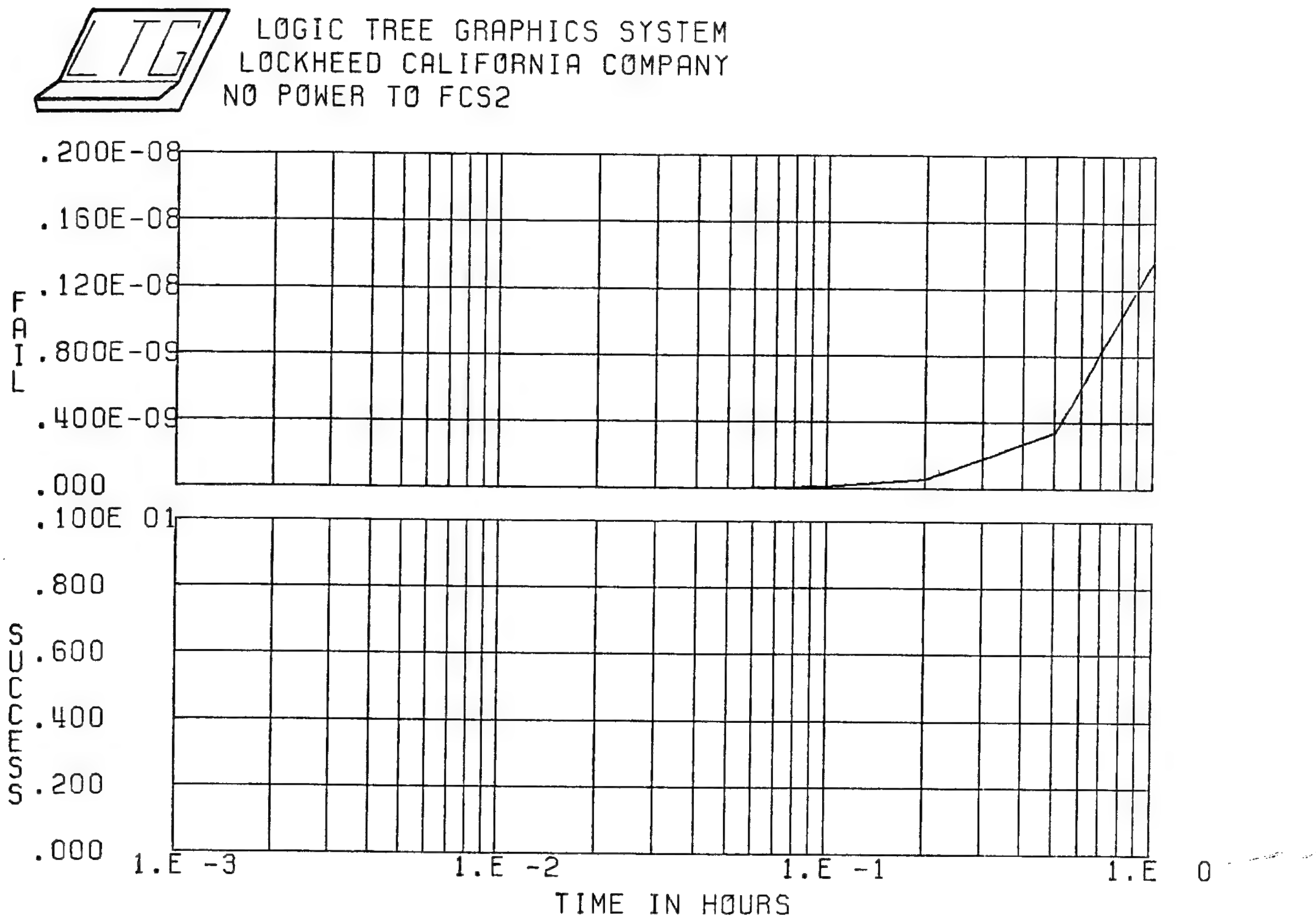
NO POWER TO FCS1 4/23/84 MAY 02, 1984
MINIMAL CUT SET DATA IN DESCENDING ORDER OF PROBABILITTY

NO.	CUT SET NO.	MAX FAILURE PROB.	COMPONENTS CONTAINED IN SET				
1	1	.13600814E-08	INV2	INV1			
2	3	.14160428E-12	APU	BT8	ENG2		
3	7	.12090941E-12	APU	BT8	GEN3		
4	2	.44258644E-13	APUC	BT8	ENG2		
5	5	.37790429E-13	APUC	BT8	GEN3		
6	8	.24727858E-14	APU	BT8	GC3A		
7	12	.10540982E-14	APU	ENG2	ENG1	ENG3	
8	6	.77287310E-15	APUC	BT8	GC3A		
9	4	.66983005E-15	GC7	BT8	ENG2		
10	9	.57193719E-15	GC7	BT8	GEN3		
11	11	.32946003E-15	APUC	ENG2	ENG1	ENG3	
12	10	.11697006E-16	GC7	BT8	GC3A		
13	61	.99899481E-19	APU	ENG2	GEN2	GEN1	ENG3
14	45	.99899481E-19	APU	GEN3	ENG1	GEN4	ENG3
15	74	.31223751E-19	APUC	GEN3	ENG1	GEN4	ENG3
16	47	.31223751E-19	APUC	ENG2	GEN2	GEN1	ENG3
17	72	.20431006E-20	APU	ENG2	GC2	GEN1	ENG3
18	60	.20431006E-20	APU	ENG2	GEN2	GC1	ENG3
19	18	.20431006E-20	APU	GEN3	ENG1	ENG3	BT2
20	57	.20431006E-20	APU	ENG2	GC12	GEN1	ENG3
21	49	.20431006E-20	APU	GEN3	GC12	ENG1	ENG3
22	24	.20431006E-20	APU	GEN3	ENG1	ENG3	GC4
23	63	.17445102E-20	APU	GEN3	GC12	GEN1	ENG3
24	76	.63857435E-21	APUC	GEN3	ENG1	ENG3	GC4
25	67	.63857435E-21	APUC	GEN3	ENG1	ENG3	BT2

Table 11. Logic Structure, No Power to FCS 2

```
*****************************************************************
******* CUTSETS OBTAINED BY MOCUS - NO POWER TO FCS2 4/23/84 *******
*****************************************************************
 101 ELEMENTS USED TO DESCRIBE THIS TREE -   40 GATES ARE DEFINED
```

GATE	TYPE	# GATES	# COMP.	INPUTS			
2	OR	2	0	22	23		
22	AND	3	0	B2	N3	B1	
B2	OR	1	1	BT2	B4		
B4	AND	2	0	B5	B6		
B5	OR	0	3	GC2	ENG1	GEN2	
B6	OR	1	1	GC1	B7		
B7	AND	3	0	B8	B16	A8	
B8	OR	0	3	GC1	ENG1	GEN1	
B16	OR	1	1	BT8	B12		
A8	OR	1	1	BT1	A11		
A11	AND	3	0	N11	A12	A13	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
A12	OR	1	1	GT56	A14		
A13	OR	0	3	GC5	ENG3	GEN5	
A14	AND	2	0	A16	A17		
A16	OR	1	1	BT3	A21		
A17	OR	0	3	GC6	ENG3	GEN6	
A21	OR	2	0	N15	N16		
N15	OR	0	3	GC4	GEN4	ENG2	
N16	OR	0	4	BT4	GC3	ENG2	GEN3
B12	AND	2	0	B21	B22		
B21	OR	0	3	GC7	APUC	APU	
B22	OR	0	3	GC3A	ENG2	GEN3	
N3	AND	2	0	N4	N5		
N4	OR	1	1	BT3	N6		
N5	OR	0	3	GC4	ENG2	GEN4	
N6	AND	2	0	N7	N8		
N7	OR	0	3	GC6	ENG3	GEN6	
N8	OR	1	1	GT56	N9		
N9	AND	2	0	N10	N11		
N10	OR	0	3	GC5	ENG3	GEN5	
N11	OR	1	1	BT7	N12		
N12	AND	2	0	N13	N14		
N13	OR	0	3	GC4A	ENG3	GEN4	
N14	OR	0	3	GC8	APUC	APU	
B1	OR	0	3	GC3A	GEN3	ENG2	
23	AND	0	2	INV1	INV2		



TIME IN HOURS	FAIL PROB.	SUCCESS PROB.
0.0	0.0	1.00000000E 00
9.99999931E-04	1.36002321E-15	9.99999940E-01
1.99999986E-03	5.44009282E-15	9.99999940E-01
4.99999896E-03	3.40005801E-14	9.99999940E-01
9.99999791E-03	1.36002321E-13	9.99999940E-01
1.99999996E-02	5.44050915E-13	9.99999940E-01
4.99999970E-02	3.40030781E-12	9.99999940E-01
9.99999642E-02	1.36012451E-11	9.99999940E-01
1.99999988E-01	5.44048556E-11	9.99999940E-01
5.00000000E-01	3.40026673E-10	9.99999940E-01
1.00000000E 00	1.36008338E-09	9.99999940E-01

Figure 8. Failure Probability - Time History No Power to FCS 2

Table 12. Cut Set, No Power to FCS 2

NO POWER TO FCS2 4/23/84 APRIL 25, 1984

MINIMAL CUT SET DATA IN DESCENDING ORDER OF PROBABILITTY

NO.	CUT SET NO.	MAX FAILURE PROB.	COMPONENTS CONTAINED IN SET				
1	1	.13600814E-08	INV1	INV2			
2	18	.10540982E-14	ENG1	ENG2	ENG3	APU	
3	2	.66983005E-15	BT2	ENG2	BT3		
4	17	.32946003E-15	ENG1	ENG2	ENG3	APUC	
5	14	.18407336E-16	BT2	ENG2	ENG3	APU	
6	15	.57532455E-17	BT2	ENG2	ENG3	APUC	
7	87	.99899481E-19	ENG1	GEN3	GEN4	ENG3	APU
8	67	.99899481E-19	GEN2	ENG2	ENG3	GEN1	APU
9	7	.87072185E-19	BT3	ENG2	ENG1	GC1	
10	16	.87072185E-19	BT2	ENG2	ENG3	GC8	
11	13	.87072185E-19	BT2	ENG2	ENG3	BT7	
12	11	.87072185E-19	BT2	ENG2	GT56	ENG3	
13	3	.74346942E-19	BT3	ENG2	GEN2	GC1	
14	6	.74346942E-19	BT2	ENG2	GT56	GEN6	
15	4	.63481459E-19	BT2	GEN3	GEN4	BT3	
16	100	.31223751E-19	ENG1	GEN3	GEN4	ENG3	APUC
17	85	.31223751E-19	GEN2	ENG2	ENG3	GEN1	APUC
18	84	.20431006E-20	ENG1	GEN3	GC4	ENG3	APU
19	80	.20431006E-20	GC2	ENG2	ENG3	GEN1	APU
20	79	.20431006E-20	ENG1	GC3A	GEN4	ENG3	APU
21	68	.20431006E-20	GEN2	ENG2	ENG3	GC1	APU
22	75	.17445102E-20	BT2	GEN3	GEN4	ENG3	APU
23	12	.15205108E-20	BT2	ENG2	GT56	GC6	
24	8	.15205108E-20	BT3	ENG2	GC2	GC1	
25	9	.12982947E-20	BT2	GEN3	GC4	BT3	

Table 13. Component Failure Rate

COMPONENT	LAMBDA	Reference
APU	0.47999970E-03	
APUC	0.14999991E-03	
BT1	0.22699978E-05	
BT2	0.22699978E-05	
BT3	0.22699978E-05	
BT4	0.22699978E-05	
BT7	0.22699978E-05	
BT8	0.22699978E-05	
C3A	0.22699978E-05	
ENG1	0.12999991E-03	
ENG2	0.12999991E-03	L1011
ENG3	0.12999991E-03	
GC1	0.22699978E-05	
GC12	0.22699978E-05	
GC2	0.22699978E-05	
GC3	0.22699978E-05	
GC3A	0.22699978E-05	
GC4	0.22699978E-05	
GC4A	0.22699978E-05	
GC5	0.22699978E-05	
GC6	0.22699978E-05	
GC7	0.22699978E-05	
GC8	0.22699978E-05	
GEN1	0.11099993E-03	
GEN2	0.11099993E-03	
GEN3	0.11099993E-03	Quoted
GEN4	0.11099993E-03	
GEN5	0.11099993E-03	
GEN6	0.11099993E-03	
GT56	0.22699978E-05	
INV1	0.36879967E-04	L1011
INV2	0.99999932E-04	

APPENDIX C

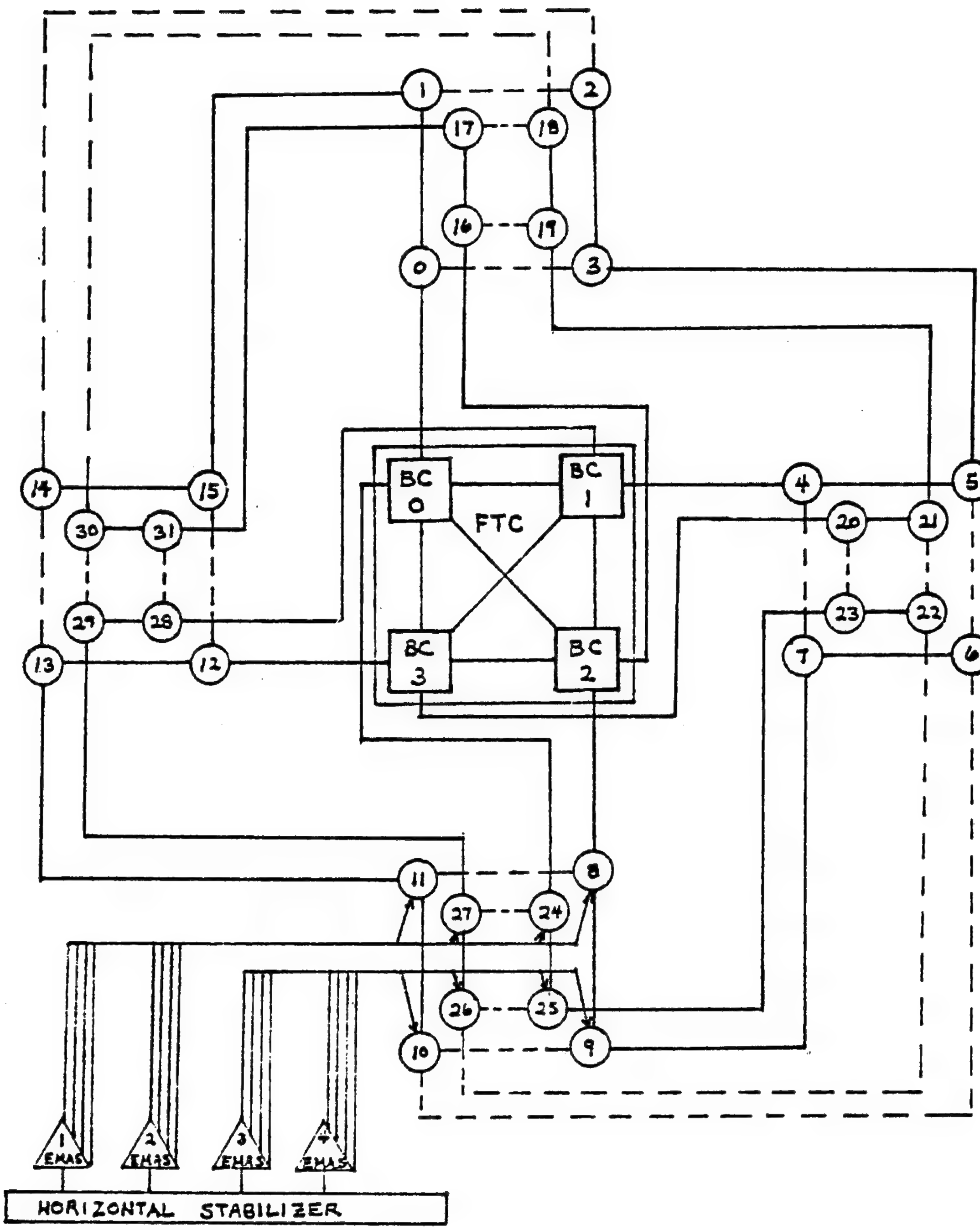
INTEGRATED DIGITAL ELECTRIC AIRCRAFT (IDEA) FLIGHT CONTROL SYSTEM RELIABILITY ANALYSIS MESH NETWORK CONFIGURATIONS

By
W.G. Ness

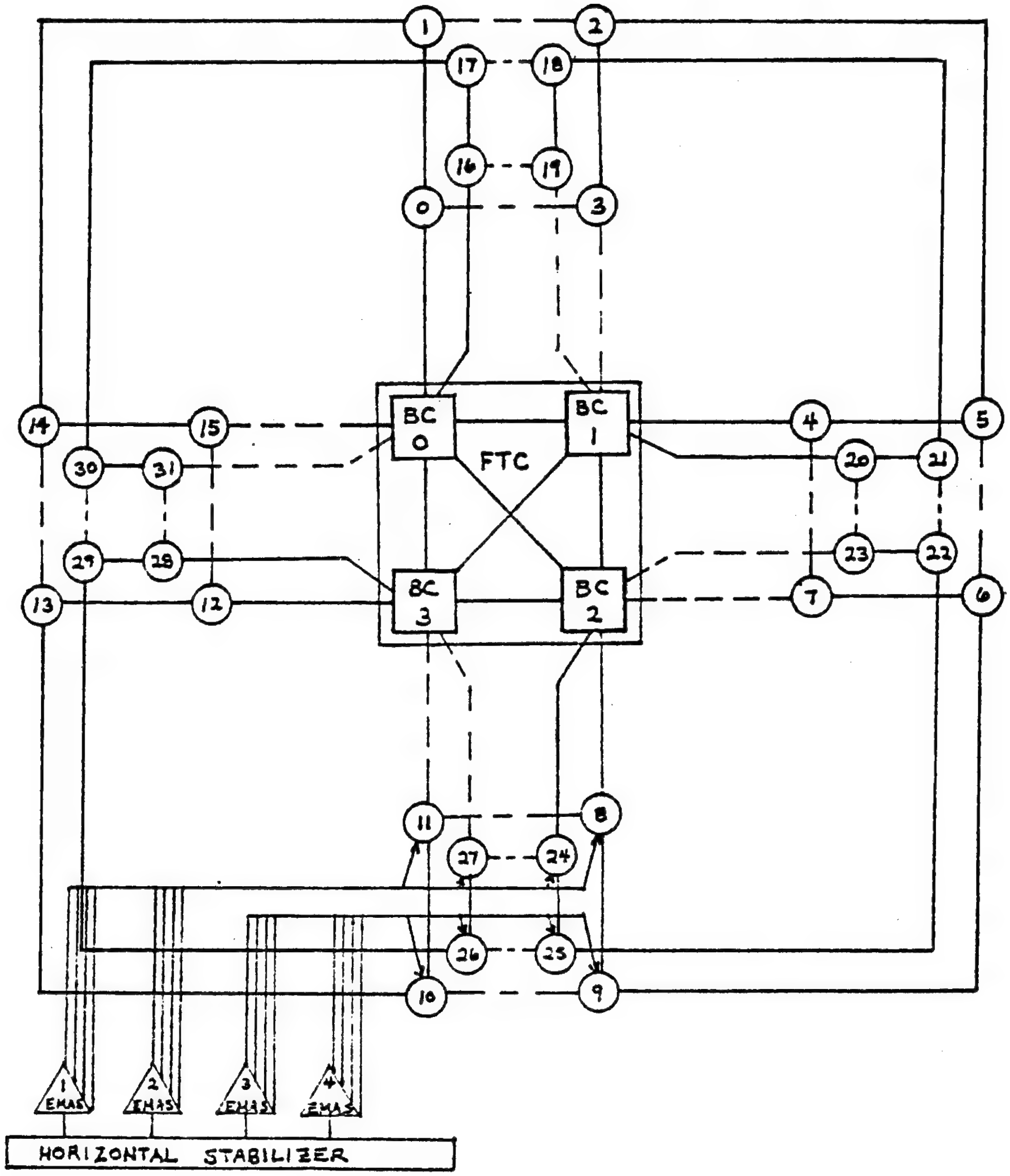
INTEGRATED DIGITAL ELECTRIC AIRCRAFT (IDEA) - NASA FLIGHT CONTROL SYSTEM RELIABILITY ANALYSIS MESH NETWORK CONFIGURATIONS

The purpose of this report is to outline the reliability analysis for the flight control system on the Integrated Digital Electric Aircraft (IDEA). The particular architectures under consideration are called "mesh network" configurations. The high redundancy of paths in these configurations is designed to produce a very survivable system. Three mesh networks are examined in this effort.

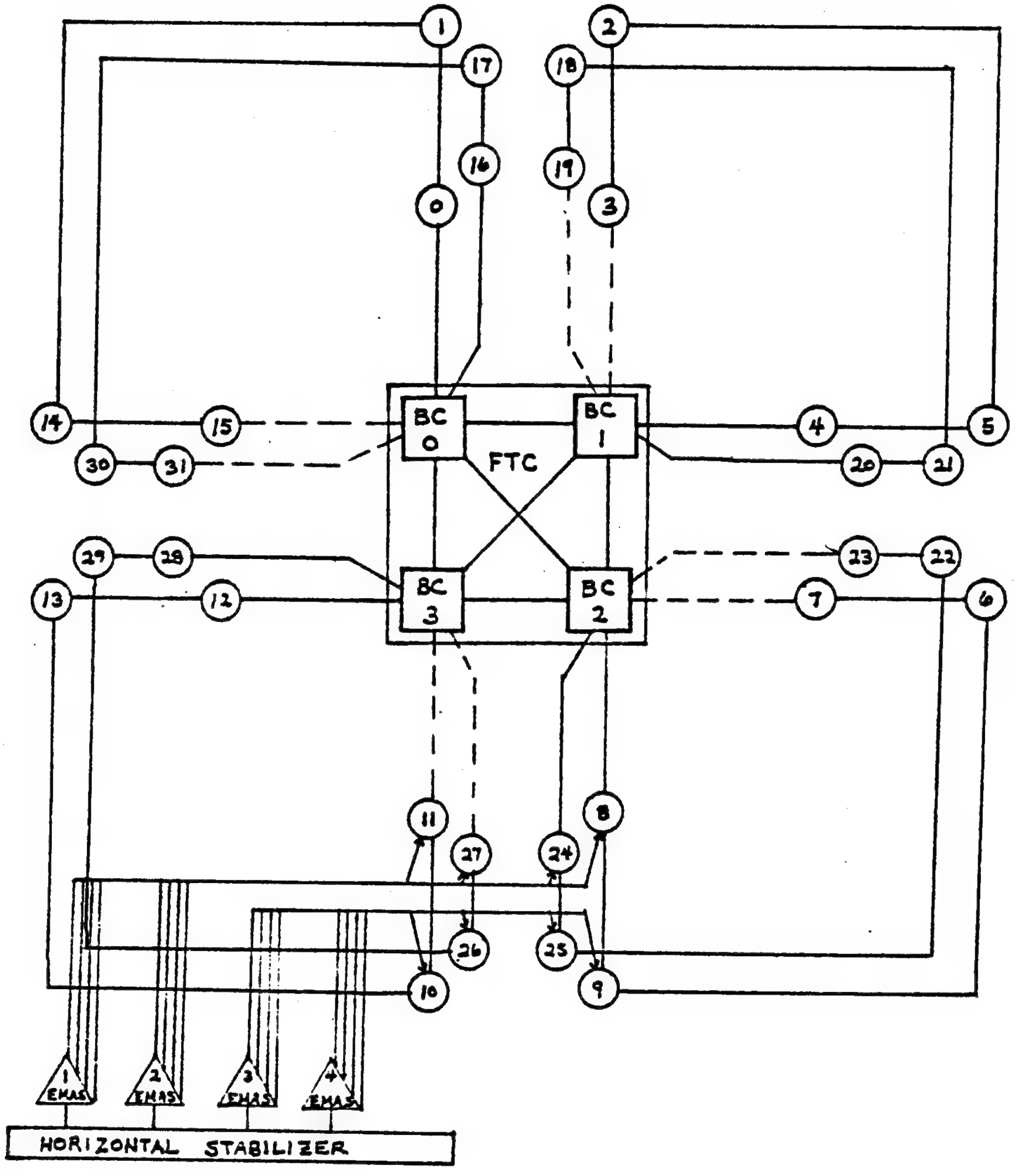
The simplified diagrams on the following pages display the differences between the three meshes. Only the elevator actuation in the horizontal stabilizer is drawn, but the analysis includes all sensors, processors, bus controllers, engine controllers, and actuators necessary for the pitch, roll, yaw, and thrust control functions. Immediately following the diagrams is a table showing the results of the analysis for various mission times.



MESH NETWORK 1



MESH NETWORK 2



MESH NETWORK 3

HRS	NETWORK 1	NETWORK 2	NETWORK 3
2.0	$.6363838 \times 10^{-11}$	$.6363838 \times 10^{-11}$	$.6363840 \times 10^{-11}$
4.0	$.1339661 \times 10^{-10}$	$.1339661 \times 10^{-10}$	$.1339664 \times 10^{-10}$
6.0	$.2192058 \times 10^{-10}$	$.2192058 \times 10^{-10}$	$.2192071 \times 10^{-10}$
8.0	$.3293710 \times 10^{-10}$	$.3293710 \times 10^{-10}$	$.3293749 \times 10^{-10}$
10.0	$.4761293 \times 10^{-10}$	$.4761293 \times 10^{-10}$	$.4761390 \times 10^{-10}$

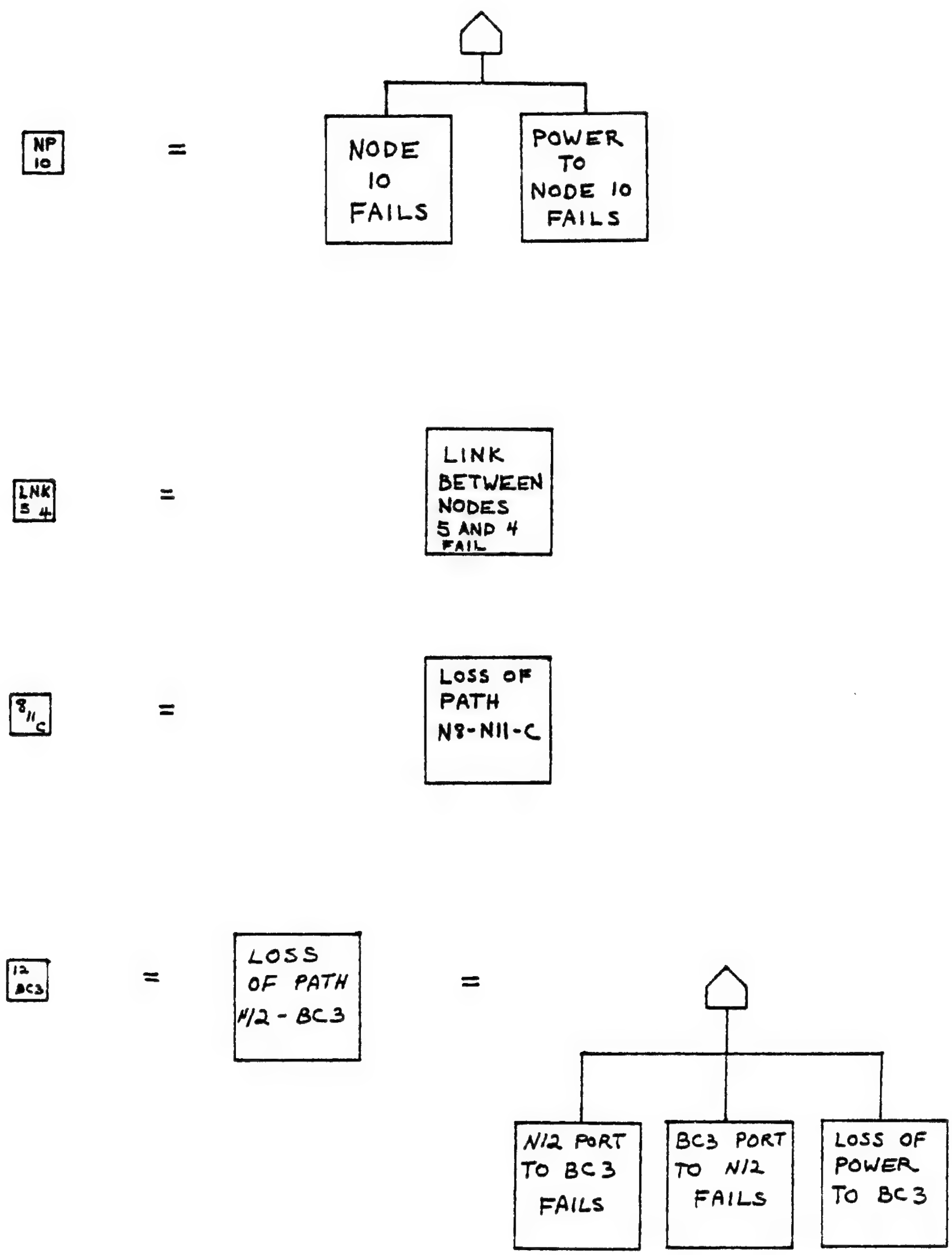
FLIGHT CONTROL SYSTEM
FAILURE PROBABILITIES

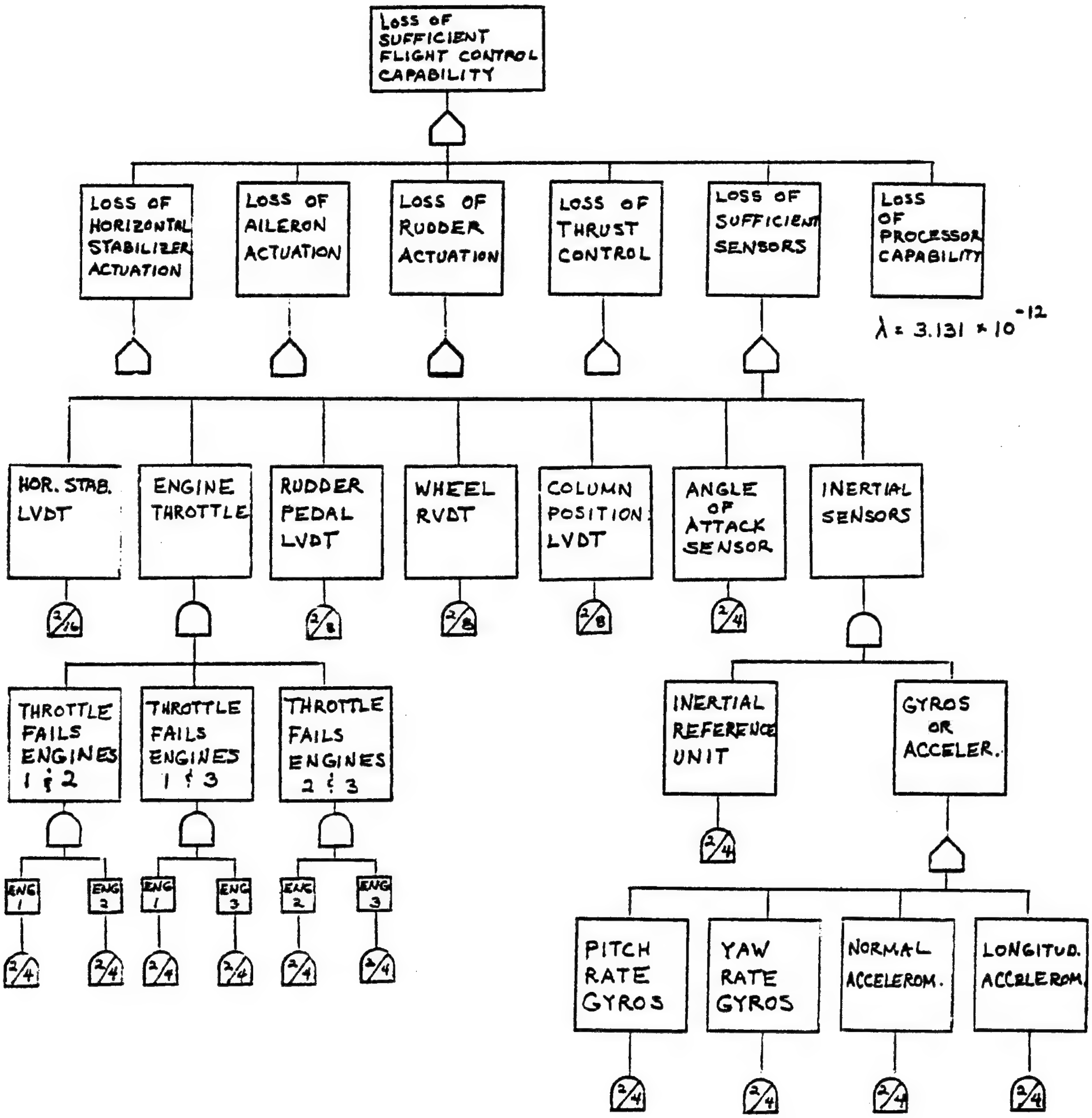
FAULT TREE ANALYSIS

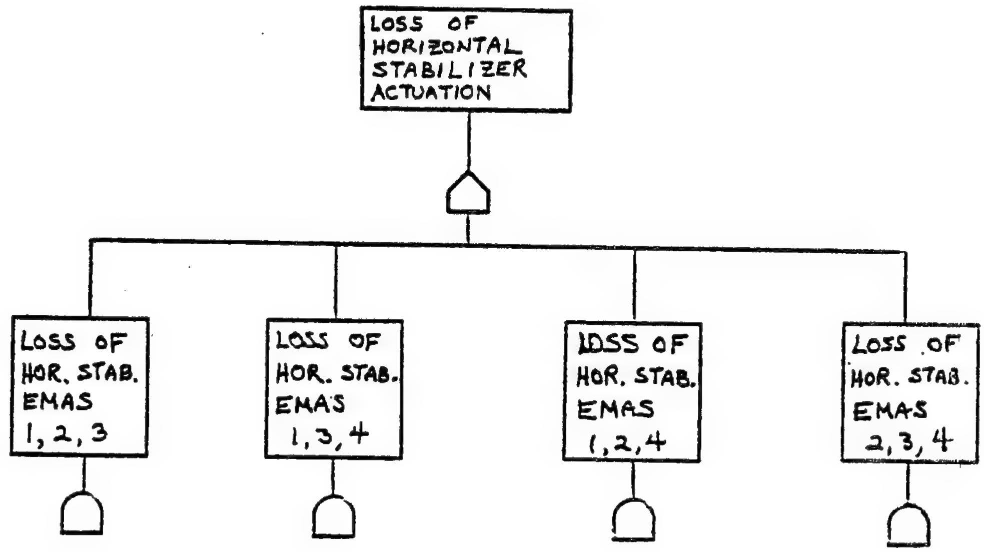
The fault tree is constructed from the top event (flight control system failure) on down to the individual failures of equipment units or communication links or power supplies. After constructing a few levels of the tree, it becomes evident that several major "branches" are repeated throughout, thus saving much unnecessary effort in recalculation. Furthermore, it will be shown that, in the mesh, probability of failure drops quickly enough to be negligible after a few levels. These two observations are important, since they permit a feasible and accurate analysis for a system whose complexity could otherwise completely prohibit reasonable predictions for system reliability.

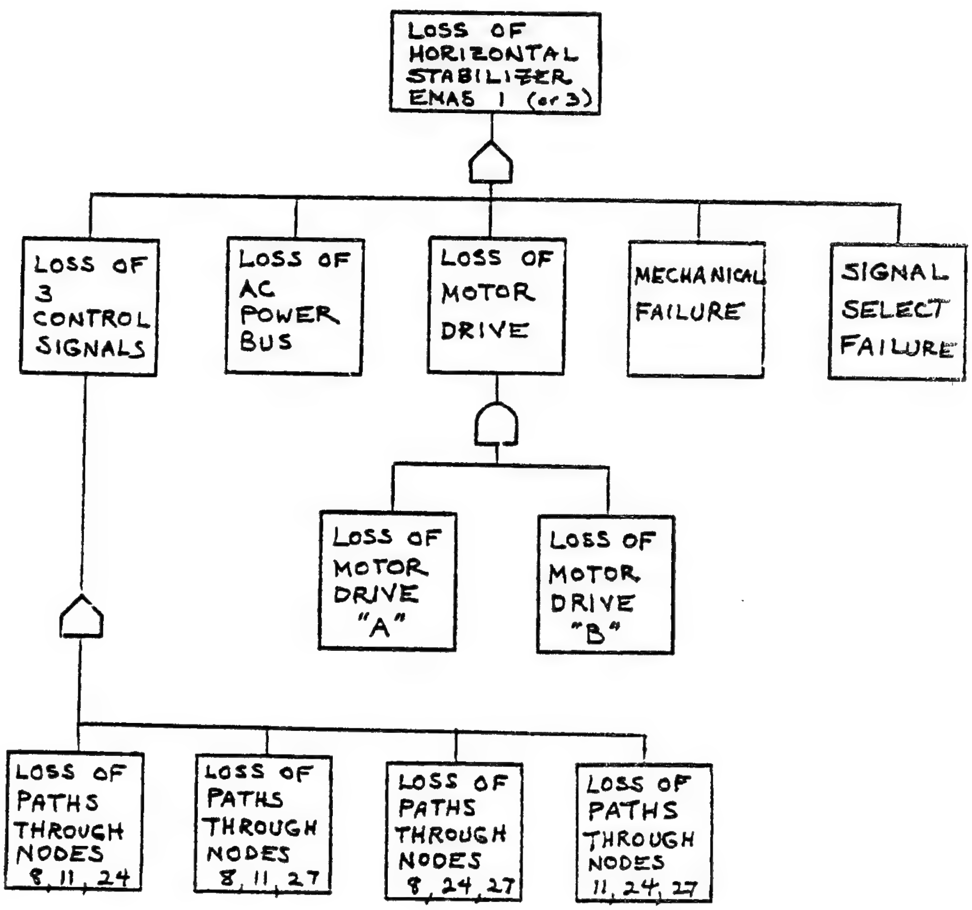
Some definitions should be helpful. A "subscriber" is an actuator, engine controller, or any flight control mechanism which requires a command from the processors. "BC2" is an abbreviation for "bus controller two". A "link" is the path from a subscriber to the first node, including the ports in the subscriber and the node.

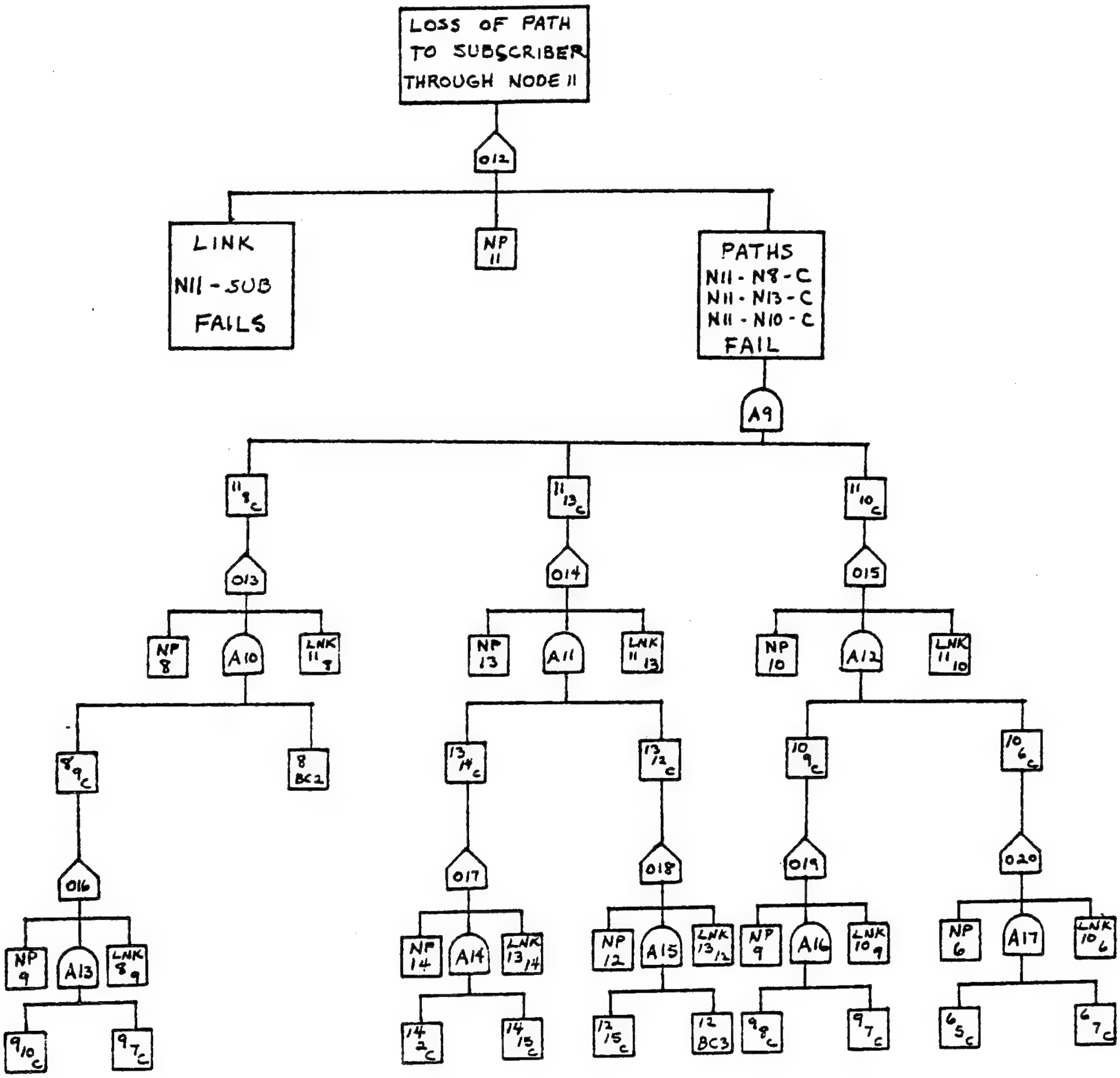
SHORTHAND SYMBOLS




LOSS OF SUFFICIENT FLIGHT CONTROL CAPABILITY
LOSS OF HORIZONTAL STABILIZER ACTUATION
LOSS OF AILERON ACTUATION
LOSS OF RUDDER ACTUATION
LOSS OF THRUST CONTROL
LOSS OF SUFFICIENT SENSORS
LOSS OF PROCESSOR CAPABILITY
λ = 3.131 × 10^-12
HOR. STAB. LVDT
ENGINE THROTTLE
RUDDER PEDAL LVDT
WHEEL RVDT
COLUMN POSITION LVDT
ANGLE OF ATTACK SENSOR
INERTIAL SENSORS
2/16
2/8
2/8
2/8
2/4
THROTTLE FAILS ENGINES 1 & 2
THROTTLE FAILS ENGINES 1 & 3
THROTTLE FAILS ENGINES 2 & 3
ENG 1
ENG 2
ENG 1
ENG 3
ENG 2
ENG 3
2/4
2/4
2/4
2/4
2/4
2/4
INERTIAL REFERENCE UNIT
GYROS OR ACCELER.
2/4
PITCH RATE GYROS
YAW RATE GYROS
NORMAL ACCELEROM.
LONGITUD. ACCELEROM.
2/4
2/4
2/4
2/4


LOSS OF HORIZONTAL STABILIZER ACTUATION
LOSS OF HOR. STAB. EMAS 1, 2, 3
LOSS OF HOR. STAB. EMAS 1, 3, 4
LOSS OF HOR. STAB. EMAS 1, 2, 4
LOSS OF HOR. STAB. EMAS 2, 3, 4


LOSS OF HORIZONTAL STABILIZER EMAS 1 (or 3)
LOSS OF 3 CONTROL SIGNALS
LOSS OF AC POWER BUS
LOSS OF MOTOR DRIVE
MECHANICAL FAILURE
SIGNAL SELECT FAILURE
LOSS OF MOTOR DRIVE "A"
LOSS OF MOTOR DRIVE "B"
LOSS OF PATHS THROUGH NODES 8, 11, 24
LOSS OF PATHS THROUGH NODES 8, 11, 27
LOSS OF PATHS THROUGH NODES 8, 24, 27
LOSS OF PATHS THROUGH NODES 11, 24, 27


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 11
O12
LINK N11 - SUB FAILS
NP 11
PATHS N11 - N8 - C N11 - N13 - C N11 - N10 - C FAIL
A9
O13
O14
O15
NP 8
A10
NP 13
A11
NP 10
A12
8 BC2
O16
O17
O18
O19
O20
NP 9
A13
NP 14
A14
NP 12
A15
NP 9
A16
NP 6
A17
12 BC3

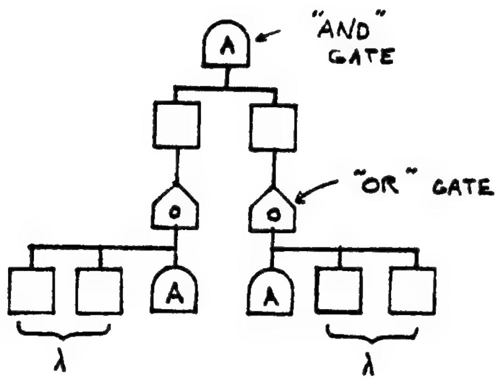
NETWORK 1

Let event E N11 be the event that the path to a subscriber through node 11 is lost. (See page)
Let $\lambda = 1 - \exp(-6.0t \times 10^{-6})$ = probability of node to bus-controller failure = probability of node + node power + link to adjacent node failure
Let $\mu = 1 - \exp(-3.0t \times 10^{-6})$ = probability of node + node power + node to subscriber link failure

$$
\begin{aligned}
P(\text{E N11} \mid \text{NO POWER FAILURE}) &= \mu + P(A9) \\
&= \mu + P(O13)P(O14)P(O15) \\
&= \mu + [\lambda + P(A10)][\lambda + P(A11)][\lambda + P(A12)] \\
&= \mu + [\lambda + \lambda^2][\lambda + \lambda^2][\lambda + \lambda^2] \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$

Two assumptions are vital to this analysis. They are:

1.) Terms of order λ^4 and higher are neglible, since $\lambda \cong -6 \times 10^{-5} \Rightarrow \lambda^4 \cong -1.296 \times 10^{-17}$
2.) The probabilities $P(A\gamma) = \lambda^2$, that is, since the fault tree for the mesh network could be infinitely recursive if drawn completely, one may truncate at an "AND" gate with a failure probability of λ^2. The reasoning for this is as follows. The general form of and "AND" gate in the fault tree for this system is as in the diagram shown below:



$$P(A) = P(o)P(o)$$
$$= [\lambda + P(A)][\lambda + P(A)]$$

or

$$z = (\lambda + z)^2$$
$$z = \lambda^2 + 2\lambda z + z^2$$
$$z^2 + (2\lambda - 1)z + \lambda^2 = 0$$

$$z = \frac{(1-2\lambda) \pm \sqrt{(2\lambda-1)^2 - 4\lambda^2}}{2}$$

$$z = \frac{(1-2\lambda) \pm \sqrt{4\lambda^2 - 4\lambda + 1 - 4\lambda^2}}{2}$$

$$z = \frac{(1-2\lambda) \pm \sqrt{1-4\lambda}}{2}$$

$\lambda = 1 - \exp(-6t \times 10^{-6}) \cong .00006$ at $t = 10$ hrs.

$$z = \frac{.99988000 - .99987999}{2} = .000000003605$$

$\lambda^2 \cong .0000000036$ which is only .14% from the true value. Hence, the assumption that $P(A) = z = \lambda^2$

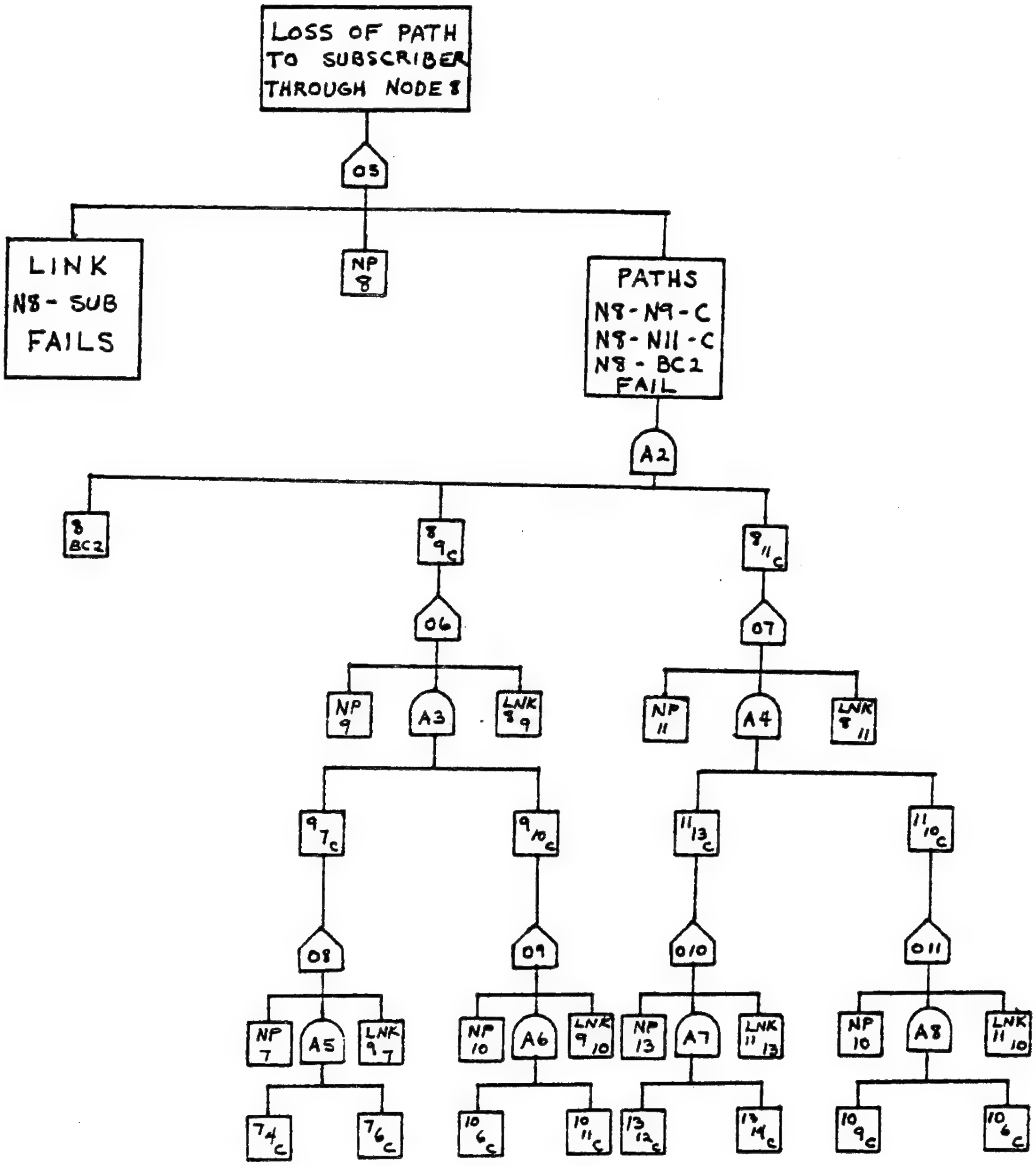
$$P(E\,N11 \mid \text{POWER SUPPLY 2 FAILS}) = \mu + P(A9)$$
$$= \mu + P(O13)\,P(O14)\,P(O15)$$
$$= \mu + [\lambda + P(A11)]$$
$$= \mu + [\lambda + P(O17)\,P(O18)]$$
$$= \mu + [\lambda + [\lambda + P(A14)][\lambda + P(A15)]]$$
$$= \mu + [\lambda + [\lambda + \lambda^2]^2]$$
$$= \mu + [\lambda + \lambda^2 + 2\lambda^3 + \lambda^4]$$
$$= \mu + \lambda + \lambda^2 + 2\lambda^3 + o(\lambda^4)$$

$$
\begin{aligned}
P(E\ N11 \mid \text{POWER SUPPLY 1 FAILS}) &= \mu + P(A9) \\
&= \mu + P(013)P(014)P(015) \\
&= \mu + [\lambda + P(A10)][\lambda + P(A11)][\lambda + P(A12)] \\
&= \mu + [\lambda + \lambda^2]^3 \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
\begin{aligned}
P(E\ N11 \mid \text{POWER SUPPLY 0 FAILS}) &= \mu + P(A9) \\
&= \mu + P(013)P(014)P(015) \\
&= \mu + [\lambda + P(A10)][\lambda + P(A11)][\lambda + P(A12)] \\
&= \mu + [\lambda + \lambda^2][\lambda + \lambda][\lambda + \lambda^2] \\
&= \mu + 2\lambda^3 + o(\lambda^4)
\end{aligned}
$$

And, of course,

$$
P(E\ N11 \mid \text{POWER SUPPLY 3 FAILS}) = 1
$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 8
O5
LINK N8 - SUB FAILS
NP 8
PATHS N8-N9-C N8-N11-C N8 - BC2 FAIL
A2
8 BC2
O6
O7
NP 9
A3
LNK 8 9
NP 11
A4
LNK 8 11
O8
O9
O10
O11
NP 7
A5
LNK 9 7
NP 10
A6
LNK 9 10
NP 13
A7
LNK 11 13
NP 10
A8
LNK 11 10

NETWORK 1

Let E N8 *be the event that the path to a subscriber through node 8 is lost (See page)*

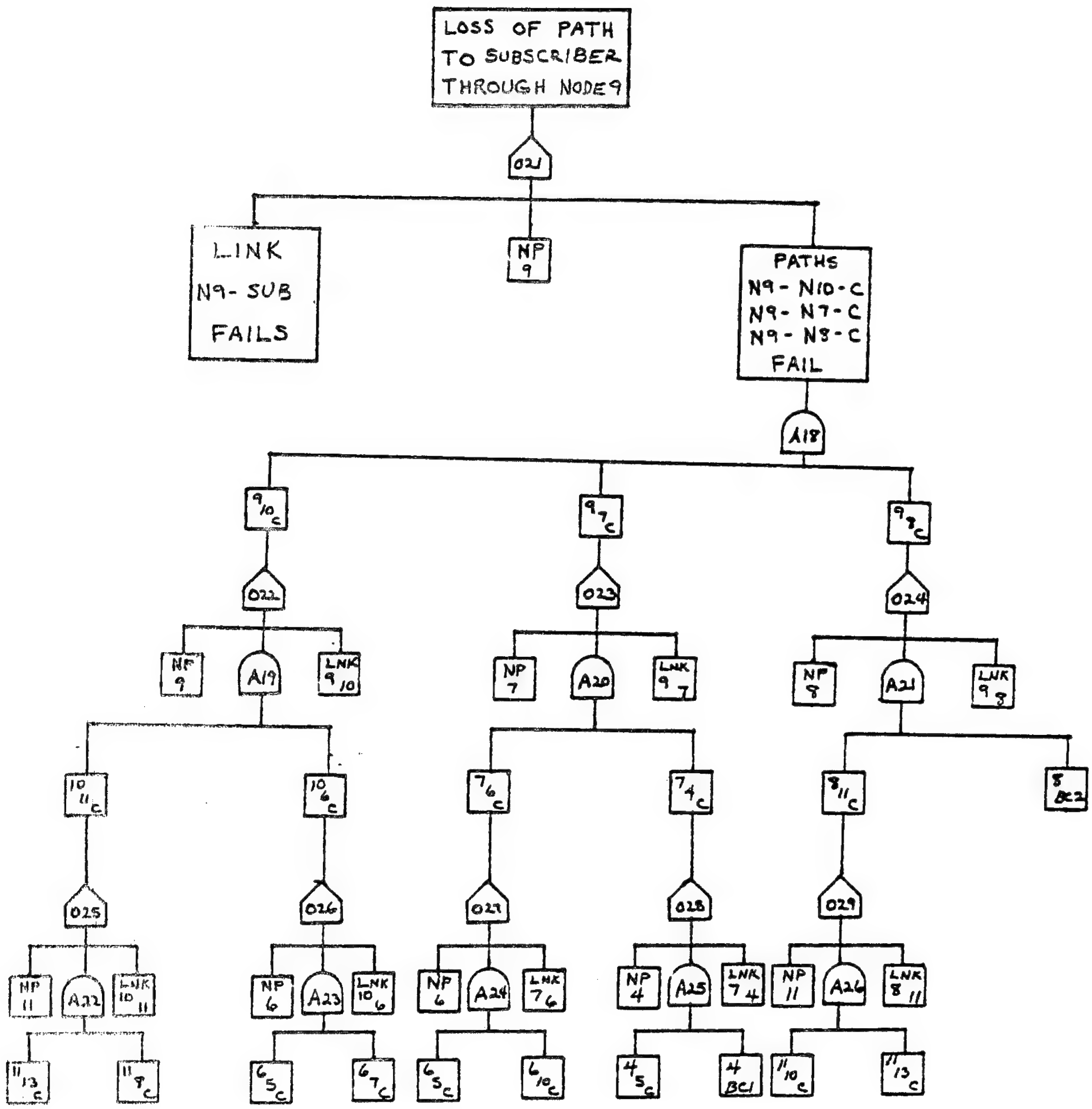
$$
\begin{aligned}
P(EN8 \mid \text{NO POWER SUPPLY LOSS}) &= \mu + P(A2) \\
&= \mu + \lambda P(06)P(07) \\
&= \mu + \lambda[\lambda + P(A3)][\lambda + P(A4)] \\
&= \mu + \lambda(\lambda + \lambda^2)^2 \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
\begin{aligned}
P(EN8 \mid \text{PS 3 FAILS}) &= \mu + P(A2) \\
&= \mu + \lambda P(06)P(07) \\
&= \mu + \lambda[\lambda + P(A3)] \\
&= \mu + \lambda[\lambda + P(08)P(09)] \\
&= \mu + \lambda[\lambda + \lambda + P(A5)] \\
&= \mu + \lambda[2\lambda + \lambda^2] \\
&= \mu + 2\lambda^2 + \lambda^3
\end{aligned}
$$

$$
P(EN8 \mid \text{PS 2 FAILS}) = 1
$$

$$
\begin{aligned}
P(EN8 \mid \text{PS 1 FAILS}) &= \mu + P(A2) \\
&= \mu + \lambda P(06)P(07) \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
\begin{aligned}
P(EN8 \mid \text{PS 0 FAILS}) &= \mu + P(A2) \\
&= \mu + \lambda P(06)P(07) \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 9
O21
LINK N9- SUB FAILS
NP 9
PATHS N9 - N10 - C N9 - N7 - C N9 - N8 - C FAIL
A18
O22
O23
O24
A19
A20
A21
O25
O26
O27
O28
O29
A22
A23
A24
A25
A26

NETWORK 1

Let E N9 be the event that the path to a subscriber through node 9 is lost. (See page)

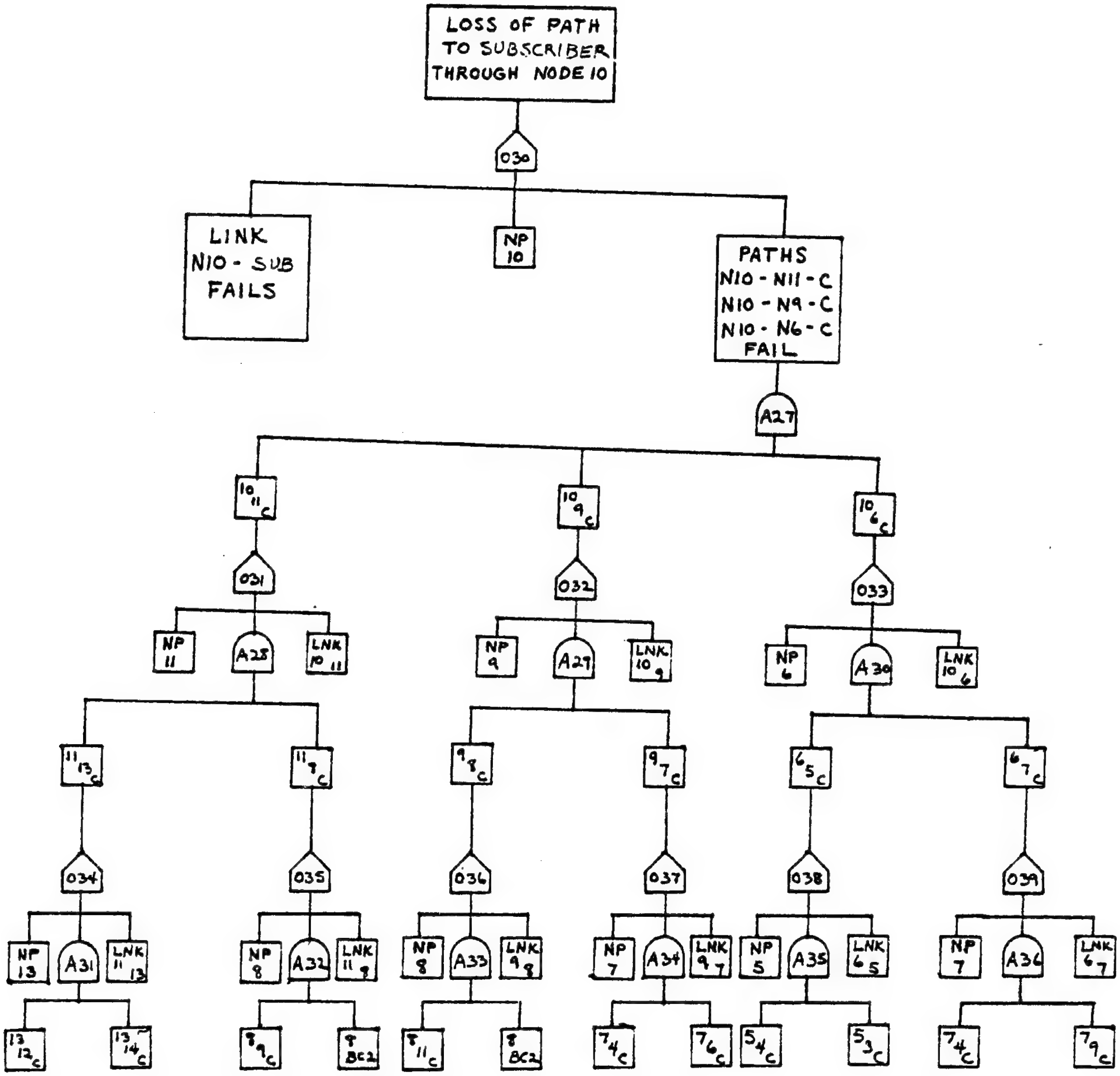
$$
\begin{aligned}
P(E\ N9 \mid \text{NO POWER SUPPLY LOSS}) &= \mu + P(A18) \\
&= \mu + P(O22)P(O23)P(O24) \\
&= \mu + [\lambda + P(A19)][\lambda + P(A20)][\lambda + P(A21)] \\
&= \mu + (\lambda + \lambda^2)^3 \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
\begin{aligned}
P(E2N9 \mid \text{PS 3 FAILS}) &= \mu + P(A18) \\
&= \mu + P(O22)P(O23)P(O24) \\
&= \mu + [\lambda + P(A20)][\lambda + P(A21)] \\
&= \mu + [\lambda + \lambda^2][\lambda + \lambda] \\
&= \mu + 2\lambda^2 + 2\lambda^3
\end{aligned}
$$

$$
P(E\ N9 \mid \text{PS 2 FAILS}) = 1
$$

$$
\begin{aligned}
P(E\ N9 \mid \text{PS 1 FAILS}) &= \mu + P(A18) \\
&= \mu + P(O22)P(O23)P(O24) \\
&= \mu + [\lambda + P(A19)][\lambda + P(A20)][\lambda + P(A21)] \\
&= \mu + [\lambda + \lambda^2][\lambda + \lambda][\lambda + \lambda^2] \\
&= \mu + 2\lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
\begin{aligned}
P(E\ N9 \mid \text{PS 0 FAILS}) &= \mu + P(A18) \\
&= \mu + P(O22)P(O23)P(O24) \\
&= \mu + [\lambda + P(A19)][\lambda + P(A20)][\lambda + P(A21)] \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 10
O30
LINK N10 - SUB FAILS
NP 10
PATHS N10 - N11 - C N10 - N9 - C N10 - N6 - C FAIL
A27
O31
O32
O33
NP 11
A28
LNK 10 11
NP 9
A29
LNK 10 9
NP 6
A30
LNK 10 6
O34
O35
O36
O37
O38
O39
NP 13
A31
LNK 11 13
NP 8
A32
LNK 11 9
NP 8
A33
LNK 9 8
NP 7
A34
LNK 9 7
NP 5
A35
LNK 6 5
NP 7
A36
LNK 6 7

NETWORK 1

Let E N10 be the event that the path to a subscriber through node 10 is lost. (See page)

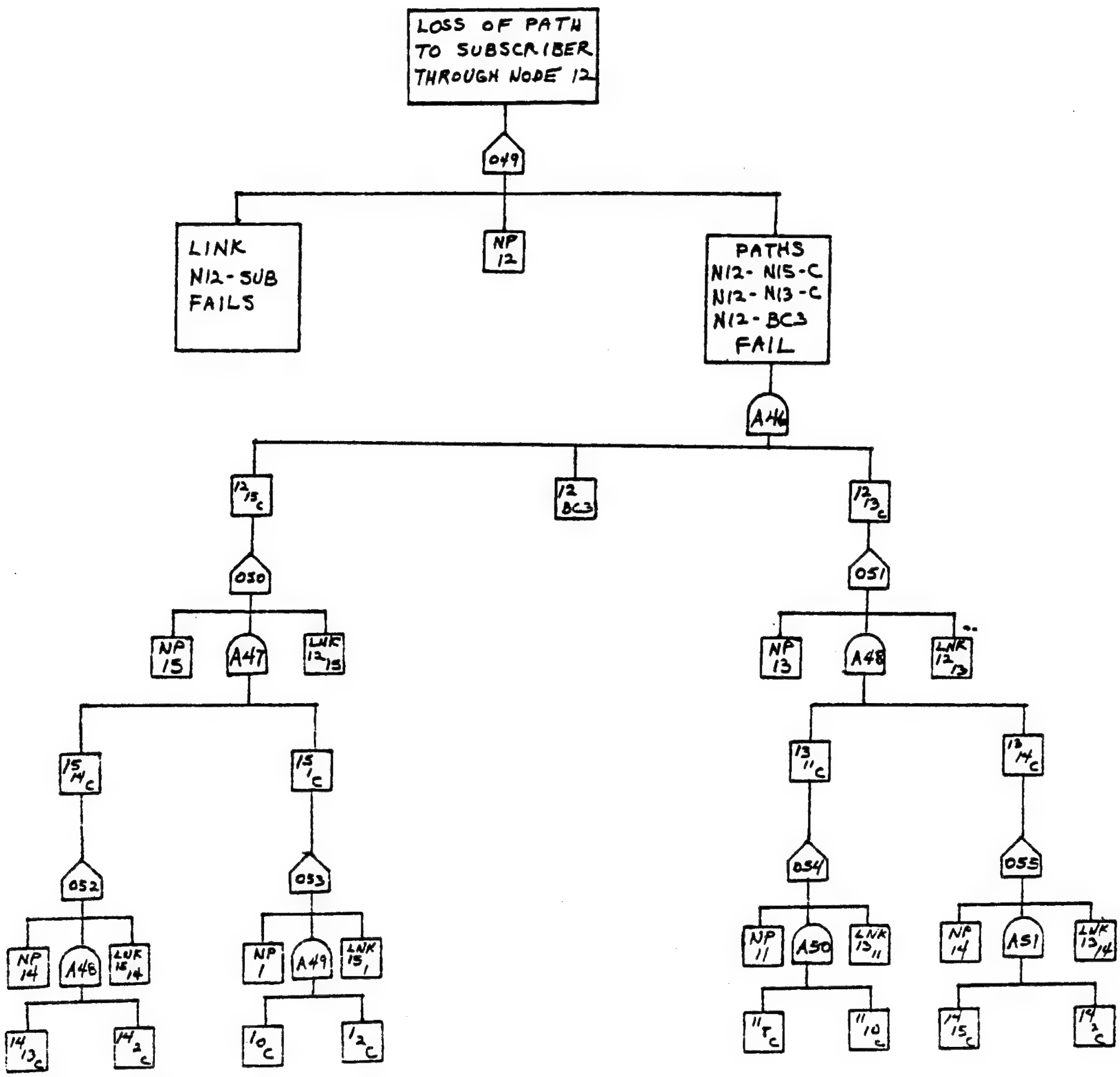
$$
\begin{aligned}
P(E\ N10 \mid \text{NO POWER SUPPLY LOSS}) &= \mu + P(A27) \\
&= \mu + P(031)\,P(032)\,P(033) \\
&= \mu + [\lambda + P(A28)][\lambda + P(A29)][\lambda + P(A30)] \\
&= \mu + (\lambda + \lambda^2)^3 \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
P(E\ N10 \mid \text{PS 3 FAILS}) = 1
$$

$$
\begin{aligned}
P(E\ N10 \mid \text{PS 2 FAILS}) &= \mu + P(A27) \\
&= \mu + P(031)\,P(032)\,P(033) \\
&= \mu + \lambda + P(A28) \\
&= \mu + \lambda + P(034)\,P(035) \\
&= \mu + \lambda + \lambda + P(A31) \\
&= \mu + 2\lambda + \lambda^2
\end{aligned}
$$

$$
\begin{aligned}
P(E\ N10 \mid \text{PS 1 FAILS}) &= \mu + P(A27) \\
&= \mu + P(031)\,P(032)\,P(033) \\
&= \mu + [\lambda + P(A28)][\lambda + P(A29)][\lambda + P(A30)] \\
&= \mu + [\lambda + \lambda^2][\lambda + \lambda^2][\lambda + P(038)\,P(039)] \\
&= \mu + [\lambda + \lambda^2]^2[\lambda + \lambda + P(A36)] \\
&= \mu + 2\lambda^3 + o(\lambda^4)
\end{aligned}
$$

$$
\begin{aligned}
P(E\ N10 \mid \text{PS 0 FAILS}) &= \mu + P(A27) \\
&= \mu + P(031)\,P(032)\,P(033) \\
&= \mu + \lambda^3 + o(\lambda^4)
\end{aligned}
$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 12
O49
LINK N12-SUB FAILS
NP 12
PATHS N12-N15-C N12-N13-C N12-BC3 FAIL
A46
12 BC3
O50
O51
NP 15
A47
LNK 12 15
NP 13
A48
LNK 12 13
O52
O53
O54
O55
NP 14
A48
LNK 15 14
NP 1
A49
LNK 15 1
NP 11
A50
LNK 13 11
NP 14
A51
LNK 13 14

NETWORK 1

Let EN12 *be the event that the path to a subscriber through node 12 is lost.*

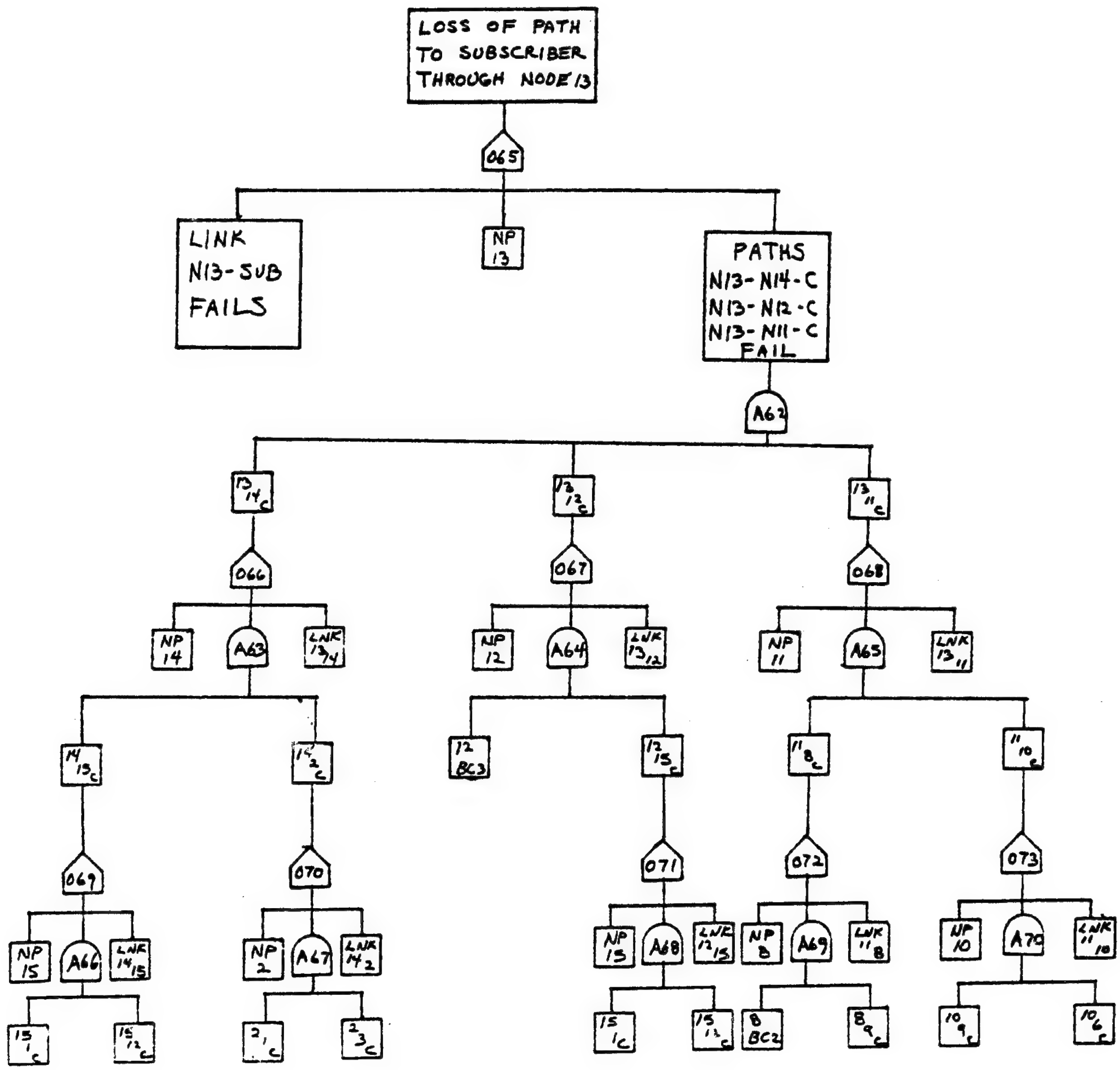
$$P(EN12 \mid \text{NO POWER SUPPLY LOSS}) = \mu + P(A46)$$
$$= \mu + P(050)\,\lambda\, P(051)$$
$$= \mu + [\lambda + P(A47)]\,\lambda\,[\lambda + P(A48)]$$
$$= \mu + [\lambda + \lambda^2]\,\lambda\,[\lambda + \lambda^2]$$
$$= \mu + \lambda^3 + o(\lambda^4)$$

$$P(EN12 \mid \text{PS 0 FAILS}) = \mu + P(A46)$$
$$= \mu + \lambda P(051)$$
$$= \mu + \lambda[\lambda + P(A48)]$$
$$= \mu + \lambda[\lambda + \lambda] = \mu + 2\lambda^2$$

$$P(EN12 \mid \text{PS 1 FAILS}) = \mu + P(A46) = \mu + \lambda P(050) P(051)$$
$$= \mu + \lambda[\lambda + \lambda^2][\lambda + \lambda^2] = \mu + \lambda^3 + o(\lambda^4)$$

$$P(EN12 \mid \text{PS 2 FAILS}) = \mu + P(A46) = \mu + \lambda P(050) P(051)$$
$$= \mu + \lambda[\lambda + P(A47)][\lambda + P(A48)]$$
$$= \mu + \lambda[\lambda + \lambda^2][\lambda + \lambda^2] = \mu + \lambda^3 + o(\lambda^4)$$

$$P(EN12 \mid \text{PS 3 FAILS}) = 1$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 13
065
LINK N13-SUB FAILS
NP 13
PATHS N13-N14-C N13-N12-C N13-N11-C FAIL
A62
066
067
068
NP 14
A63
NP 12
A64
NP 11
A65
069
070
071
072
073
NP 15
A66
NP 2
A67
A68
A69
NP 10
A70

NETWORK 1

Let ENI3 *be the event that the path to a subscriber through node 13 is lost.*

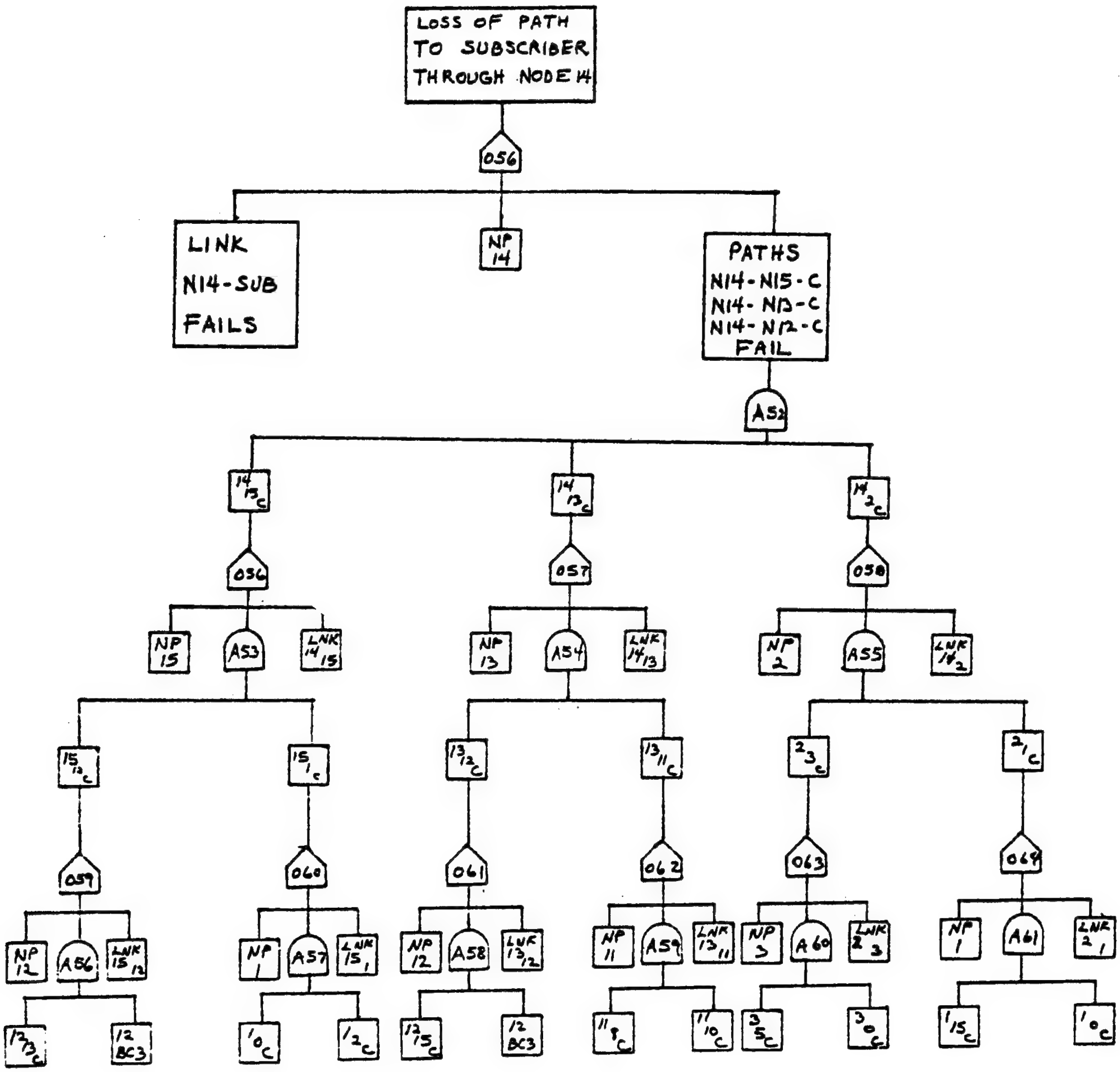
$$P(EN13 \mid \text{NO PWR SUPPLY LOSS}) = \mu + P(A62) = \mu + P(O66)P(O67)P(O68)$$
$$= \mu + [\lambda + P(A63)][\lambda + P(A64)][\lambda + P(A65)]$$
$$= \mu + [\lambda + \lambda^2]^3 = \mu + \lambda^3 + o(\lambda^4)$$

$$P(EN13 \mid \text{PS 0 FAILS}) = \mu + P(A62) = \mu + P(O67)P(O68)$$
$$= \mu + [\lambda + P(A64)][\lambda + P(A65)]$$
$$= \mu + [\lambda + \lambda^2][\lambda + \lambda^2]$$
$$= \mu + 2\lambda^2 + 2\lambda^3$$

$$P(EN13 \mid \text{PS 1 FAILS}) = \mu + P(A62) = \mu + P(O66)P(O67)P(O68)$$
$$= \mu + [\lambda + P(A63)][\lambda + P(A64)][\lambda + P(A65)]$$
$$= \mu + [\lambda + \lambda][\lambda + \lambda^2][\lambda + \lambda^2]$$
$$= \mu + 2\lambda^3 + o(\lambda^4)$$

$$P(EN13 \mid \text{PS 2 FAILS}) = \mu + P(A62) = \mu + P(O66)P(O67)P(O68)$$
$$= \mu + [\lambda + P(A63)][\lambda + P(A64)][\lambda + P(A65)]$$
$$= \mu + [\lambda + \lambda^2][\lambda + \lambda^2][\lambda + \lambda] = \mu + 2\lambda^3 + o(\lambda^4)$$

$$P(EN13 \mid \text{PS 3 FAILS}) = 1$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 14
056
LINK N14-SUB FAILS
NP 14
PATHS N14-N15-C N14-N13-C N14-N12-C FAIL
A52
036
057
058
NP 15
A53
NP 13
A54
NP 2
A55
059
060
061
062
063
064
NP 12
A56
NP 1
A57
A58
NP 11
A59
NP 3
A60
A61

NETWORK 1

Let EN14 *be the event that the path to a subscriber through node 14 is lost.*

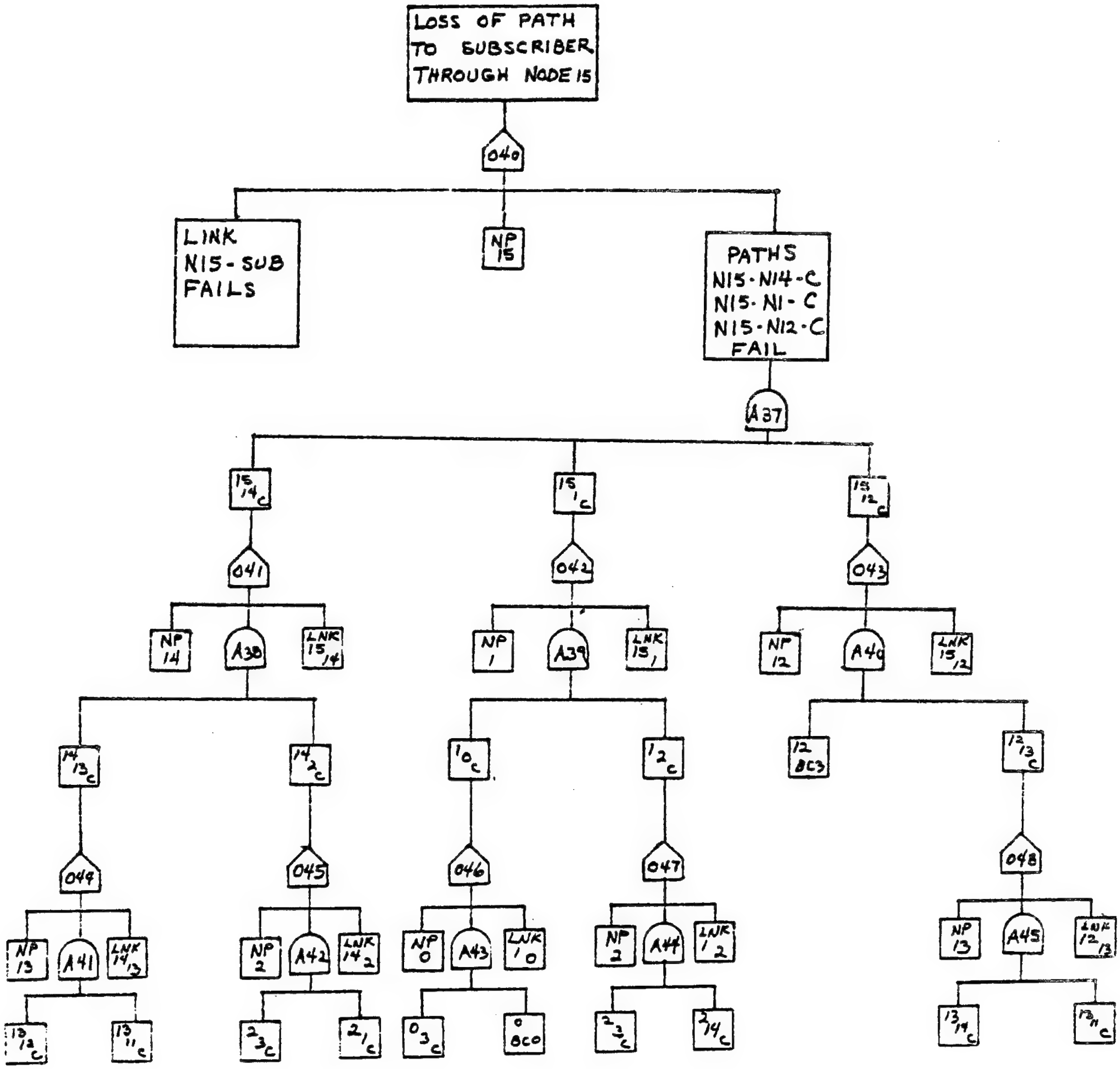
$$P(\text{EN14} \mid \text{NO PWR SUPPLY LOSS}) = \mu + P(\text{A52}) = \mu + P(\text{056})P(\text{057})P(\text{058})$$
$$= \mu + [\lambda + P(\text{A53})][\lambda + P(\text{A54})][\lambda + P(\text{A55})]$$
$$= \mu + [\lambda + \lambda^2]^3 = \mu + \lambda^3 + o(\lambda^4)$$

$$P(\text{EN14} \mid \text{PS 0 FAILS}) = 1$$

$$P(\text{EN14} \mid \text{PS 1 FAILS}) = \mu + P(\text{A52}) = \mu + P(\text{056})P(\text{057})$$
$$= \mu + [\lambda + P(\text{A53})][\lambda + P(\text{A54})]$$
$$= \mu + [\lambda + \lambda^2]\,[\lambda + \lambda^2] = \mu + \lambda^2 + 2\lambda^3 + o(\lambda^4)$$

$$P(\text{EN14} \mid \text{PS 2 FAILS}) = \mu + P(\text{A52}) = \mu + P(\text{056})P(\text{057})P(\text{058})$$
$$= \mu + [\lambda + P(\text{A53})][\lambda + P(\text{A54})][\lambda + P(\text{A55})]$$
$$= \mu + [\lambda + \lambda^2]^3 = \mu + \lambda^3 + o(\lambda^4)$$

$$P(\text{EN14} \mid \text{PS 3 FAILS}) = \mu + P(\text{A52}) = \mu + P(\text{056})P(\text{058})$$
$$= \mu + [\lambda + P(\text{A53})][\lambda + P(\text{A55})]$$
$$= \mu + [\lambda + \lambda][\lambda + \lambda^2]$$
$$= \mu + 2\lambda^2 + 2\lambda^3$$


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 15
040
LINK N15-SUB FAILS
NP 15
PATHS N15-N14-C N15-N1-C N15-N12-C FAIL
A37
041
042
043
NP 14
A38
LNK 15 14
NP 1
A39
LNK 15 1
NP 12
A40
LNK 15 12
12 BC3
044
045
046
047
048
NP 13
A41
LNK 14 13
NP 2
A42
LNK 14 2
NP 0
A43
LNK 1 0
NP 2
A44
LNK 1 2
NP 13
A45
LNK 12 13
0 BC0

NETWORK 1

Let EN15 be the event that the path to a subscriber through node 15 is lost.

$$P(\text{EN15} \mid \text{NO PWR SUPPLY LOSS}) = \mu + P(A37) = \mu + P(041)P(042)P(043)$$
$$= \mu + [\lambda + P(A38)][\lambda + P(A39)][\lambda + P(A40)]$$
$$= \mu + [\lambda + \lambda^2]^3 = \mu + \lambda^3 + o(\lambda^4)$$

$$P(\text{EN15} \mid \text{PS 0 FAILS}) = 1$$

$$P(\text{EN15} \mid \text{PS 1 FAILS}) = \mu + P(A37) = \mu + P(041)P(042)P(043)$$
$$= \mu + [\lambda + P(A38)][\lambda + P(A39)][\lambda + P(A40)]$$
$$= \mu + [\lambda + \lambda][\lambda + \lambda][\lambda + \lambda^2]$$
$$= \mu + 4\lambda^2 + 4\lambda^3$$

$$P(\text{EN15} \mid \text{PS 2 FAILS}) = \mu + P(A37) = \mu + P(041)P(042)P(043)$$
$$= \mu + [\lambda + P(A38)][\lambda + P(A39)][\lambda + P(A40)]$$
$$= \mu + [\lambda + \lambda^2]^3$$
$$= \mu + \lambda^2 + o(\lambda^4)$$

$$P(\text{EN15} \mid \text{PS 3 FAILS}) = \mu + P(A37) = \mu + P(041)P(042)$$
$$= \mu + [\lambda + P(A38)][\lambda + P(A39)]$$
$$= \mu + [\lambda + \lambda][\lambda + \lambda^2]$$
$$= \mu + 2\lambda^2 + 2\lambda^3$$

Let $\theta = 1 - \exp(-1.6t \times 10^{-8}) =$ *probability of a power supply failure*

$$
\begin{aligned}
P(E\ N11) = {} & P(E\ N11 \mid \text{NO PS FAILS})\, P(\text{NO PS FAILS}) \\
& + P(E\ N11 \mid \text{PS 3 FAILS})\, P(\text{PS 3 FAILS}) \\
& + P(E\ N11 \mid \text{PS 2 FAILS})\, P(\text{PS 2 FAILS}) \\
& + P(E\ N11 \mid \text{PS 1 FAILS})\, P(\text{PS 1 FAILS}) \\
& + P(E\ N11 \mid \text{PS 0 FAILS})\, P(\text{PS 0 FAILS})
\end{aligned}
$$

$$
\begin{aligned}
= {} & (\mu + \lambda^3)(1 - 4\theta) \\
& + 1\,\theta \\
& + (\mu + \lambda + \lambda^2 + 2\lambda^3)\theta \\
& + (\mu + \lambda^3)\theta \\
& + (\mu + 2\lambda^3)\theta \quad = W \qquad (W = P(E\,N27)\text{, also})
\end{aligned}
$$

Similarly,

$$
\begin{aligned}
P(E\ N8) = P(E\ N24) = X \\
= (\mu + \lambda^3)(1 - 4\theta) + (\mu + 2\lambda^2 + \lambda^3)\theta + (1)\theta \\
+ (\mu + \lambda^3)\theta + (\mu + \lambda^3)\theta
\end{aligned}
$$

$$
\begin{aligned}
P(E\ N9) = P(E\ N25) = Y \\
= (\mu + \lambda^3)(1 - 4\theta) + (\mu + 2\lambda^2 + 2\lambda^3)\theta + (1)\theta \\
+ (\mu + 2\lambda^3)\theta + (\mu + \lambda^3)\theta
\end{aligned}
$$

$$
\begin{aligned}
P(E\ N10) = P(E\ N26) = Z \\
= (\mu + \lambda^3)(1 - 4\theta) + (1)\theta + (\mu + 2\lambda + \lambda^2)\theta \\
+ (\mu + 2\lambda^3)\theta + (\mu + \lambda^3)\theta
\end{aligned}
$$

$$P(EN12) = (\mu+\lambda^3)(1-4\theta) + (\mu+2\lambda^2)\theta + (\mu+\lambda^3)\theta + (\mu+\lambda^3)\theta + \theta$$

$$P(EN13) = (\mu+\lambda^3)(1-4\theta) + (\mu+2\lambda^2+2\lambda^3)\theta + (\mu+2\lambda^3)\theta + (\mu+2\lambda^3)\theta + \theta$$

$$P(EN14) = (\mu+\lambda^3)(1-4\theta) + \theta + (\mu+\lambda^2+2\lambda^3)\theta + (\mu+\lambda^3)\theta + (\mu+2\lambda^2+2\lambda^3)\theta$$

$$P(EN15) = (\mu+\lambda^3)(1-4\theta) + \theta + (\mu+4\lambda^2+4\lambda^3)\theta + (\mu+\lambda^3)\theta + (\mu+2\lambda^2+2\lambda^3)\theta$$

Simplifying these expressions and consolidating terms, and noticing the symmetry of the configuration, gives the following probabilities:

$$P(EN8,24,\emptyset,16) = \mu+\theta+\lambda^3-\mu\theta-\theta\lambda^3+2\theta\lambda^2 = xx$$
$$P(EN9,25,1,17) = \mu+\theta+\lambda^3-\mu\theta+\theta\lambda^3+2\theta\lambda^2 = yy$$
$$P(EN10,26,2,18) = \mu+\theta+\lambda^3-\mu\theta-\theta\lambda^3+2\theta\lambda+\theta\lambda^2 = zz$$
$$P(EN11,27,3,19) = \mu+\theta+\lambda^3-\mu\theta+\theta\lambda^3+\theta\lambda^2+\theta\lambda = ww$$
$$P(EN12,28,4,20) = \mu+\theta+\lambda^3-\mu\theta-2\theta\lambda^3+2\theta\lambda^2 = ss$$
$$P(EN13,29,5,21) = \mu+\theta+\lambda^3-\mu\theta+2\theta\lambda^3+2\theta\lambda^2 = tt$$
$$P(EN14,30,6,22) = \mu+\theta+\lambda^3-\mu\theta+\theta\lambda^3+3\theta\lambda^2 = uu$$
$$P(EN15,31,7,23) = \mu+\theta+\lambda^3-\mu\theta+3\theta\lambda^3+6\theta\lambda^2 = vv$$

The values of these probabilities are so dominated by the terms μ and θ that, for all practical purposes they are equal. For example,

computing ww, xx, yy, zz, ss, tt, uu, and vv for mesh network #1, all values come out to be 3.016×10^{-5}. Thus, to a high degree, system failure probability is node-independent. When calculating failure probabilities for the sensors, and average value for network failure is added to component failure probability.

Having obtained an expression for probability of path loss through a given node, one must now find the probability of losing any three out of four node-paths to which a subscriber is linked. Consider the paths which supply control signals to the horizontal stabilizer EMAS.

Now, let events A, B, C, and D be such that

A = loss of paths through nodes 8, 11, 24
B = loss of paths through nodes 8, 11, 27
C = loss of paths through nodes 8, 24, 27
D = loss of paths through nodes 11, 24, 27

P[loss of 3 control signals to EMAS 1 (or EMAS 3)]

$$
\begin{aligned}
&= P[A \cup B \cup C \cup D] = P(A) + P(B) + P(C) + P(D) \\
&\quad - P(A \cap B) - P(A \cap C) - P(A \cap D) - P(B \cap C) - P(B \cap D) - P(C \cap D) \\
&\quad + P(A \cap B \cap C) + P(A \cap B \cap D) + P(A \cap C \cap D) + P(B \cap C \cap D) \\
&\quad - P(A \cap B \cap C \cap D)
\end{aligned}
$$

$$
\begin{aligned}
&= P(8)P(11)P(24) + P(8)P(11)P(27) + P(8)P(24)P(27) + P(11)P(24)P(27) \\
&\quad - 6P(8)P(11)P(24)P(27) + 4P(8)P(11)P(24)P(27) - P(8)P(11)P(24)P(27)
\end{aligned}
$$

$$= (XX)^2(WW) + (XX)(WW)^2 + (XX)^2(WW) + (XX)(WW)^2 - 6(XX)^2(WW)^2 + 4(XX)^2(WW)^2 - (XX)^2(WW)^2$$

$$= 2(XX)^2(WW) + 2(XX)(WW)^2 - 3(XX)^2(WW)^2$$

Now consider failures ___not___ due to control signal loss. By standard reliability prediction methods,

$$P[\text{loss of AC power bus}] = 1 - \exp(10t \times 10^{-6}) = D$$

$$P[\text{loss of motor drive unit}] = 1 - \exp(96t \times 10^{-6}) = E$$

$$P[\text{mechanical EMAS failure}] = 1 - \exp(2.3t \times 10^{-6}) = F$$

$$P[\text{signal select failure}] = 1 - \exp(.75t \times 10^{-6}) = G$$

$$P[\text{loss of horizontal stabilizer EMAS 1}] = P[\text{loss of hor. stab. EMAS 3}]$$
$$= 2(xx)^2(ww) + 2(xx)(ww)^2 - 3(xx)^2(ww)^2 + D + E + F + G = S = P(1) = P(3)$$

$$P[\text{loss of horizontal stabilizer EMAS 2}] = P[\text{loss of hor. stab. EMAS 4}]$$
$$= 2(yy)^2(zz) + 2(yy)(zz)^2 - 3(yy)^2(zz)^2 + D + E + F + G = T = P(2) = P(4)$$

The next step is to calculate the probability of losing the entire horizontal stabilizer actuator stage, which involves the loss of three of the four actuators.

Now let events A', B', C', D' be such that

A' = loss of horizontal stabilizer EMAS 1, 2, 3

B' = loss of horizontal stabilizer EMAS 1, 2, 4

C' = loss of horizontal stabilizer EMAS 1, 3, 4

D' = loss of horizontal stabilizer EMAS 2, 3, 4

$$P[\text{loss of 3 hor. stab. EMAS}] = P[A' \cup B' \cup C' \cup D']$$

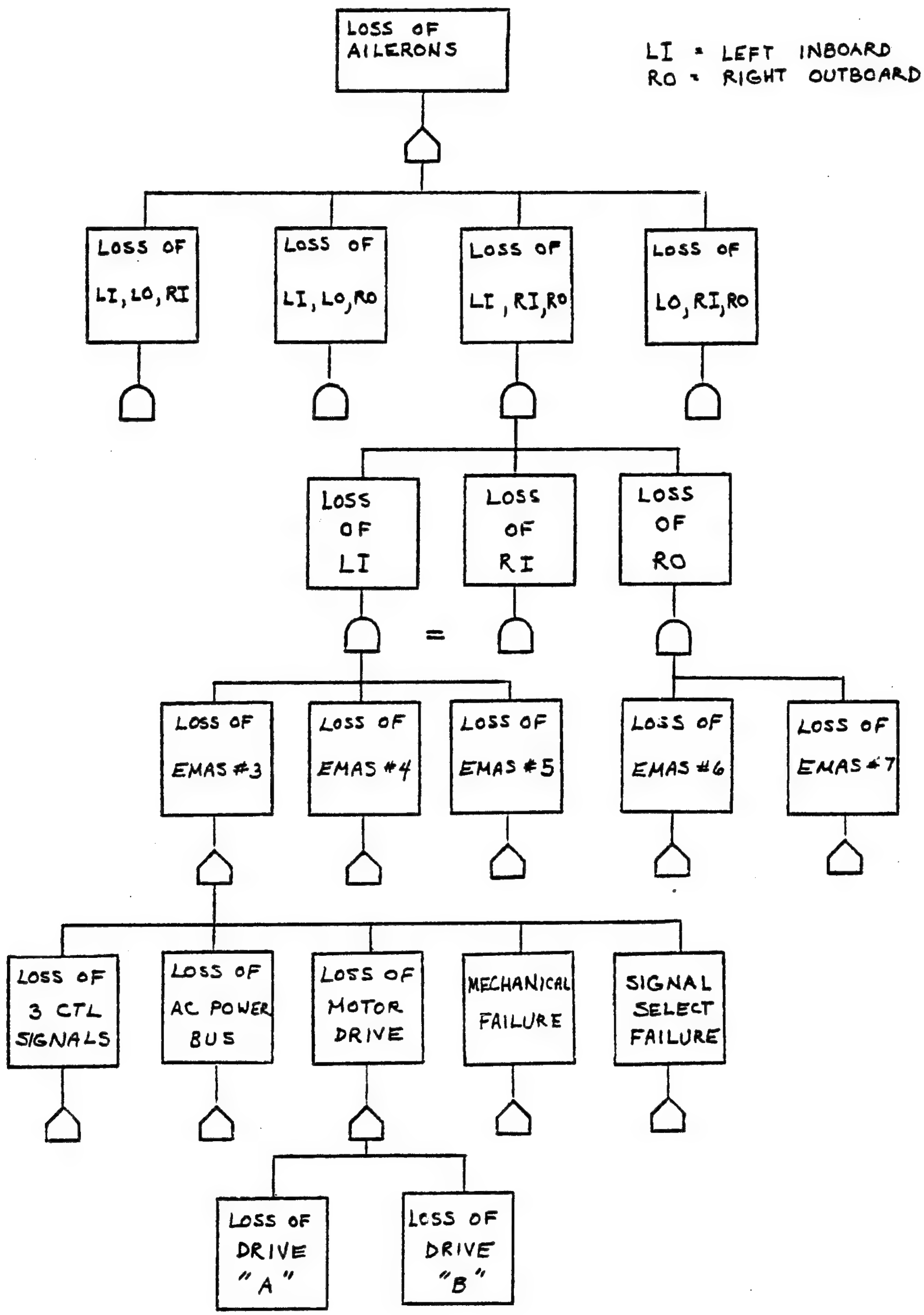
$$\begin{aligned}
&= P(A') + P(B') + P(C') + P(D') \\
&\quad - P(A' \cap B') - P(A' \cap C') - P(A' \cap D') - P(B' \cap C') - P(B' \cap D') - P(C' \cap D') \\
&\quad + P(A' \cap B' \cap C') + P(A' \cap B' \cap D') + P(A' \cap C' \cap D') + P(B' \cap C' \cap D') \\
&\quad - P(A' \cap B' \cap C' \cap D') \qquad \text{(See page)}
\end{aligned}$$

$$\begin{aligned}
&= P(1)P(2)P(3) + P(1)P(2)P(4) + P(1)P(3)P(4) + P(2)P(3)P(4) \\
&\quad - 6\,P(1)P(2)P(3)P(4) + 4\,P(1)P(2)P(3)P(4) - P(1)P(2)P(3)P(4)
\end{aligned}$$

$$= S^2T + ST^2 + S^2T + ST^2 - 6S^2T^2 + 4S^2T^2 - S^2T^2$$

$$= 2S^2T + 2ST^2 - 3S^2T^2$$

This is the probability of losing horizontal stabilizer actuation. The same basic method will be used to determine loss of ailerons, rudder, and engine controller.


LOSS OF AILERONS
LI = LEFT INBOARD
RO = RIGHT OUTBOARD
LOSS OF LI, LO, RI
LOSS OF LI, LO, RO
LOSS OF LI, RI, RO
LOSS OF LO, RI, RO
LOSS OF LI
LOSS OF RI
LOSS OF RO
=
LOSS OF EMAS #3
LOSS OF EMAS #4
LOSS OF EMAS #5
LOSS OF EMAS #6
LOSS OF EMAS #7
LOSS OF 3 CTL SIGNALS
LOSS OF AC POWER BUS
LOSS OF MOTOR DRIVE
MECHANICAL FAILURE
SIGNAL SELECT FAILURE
LOSS OF DRIVE "A"
LOSS OF DRIVE "B"

It is easy to notice that when the tree is at the level "LOSS OF EMAS #3", the analysis is exactly the same as the case of the horizontal stabilizer EMAS. Above that level, however, the greater redundancy of the roll control function is apparent. For the ailerons, as long as one actuator is operative, the surface is not failed. The loss of one EMAS is given by the following equation:

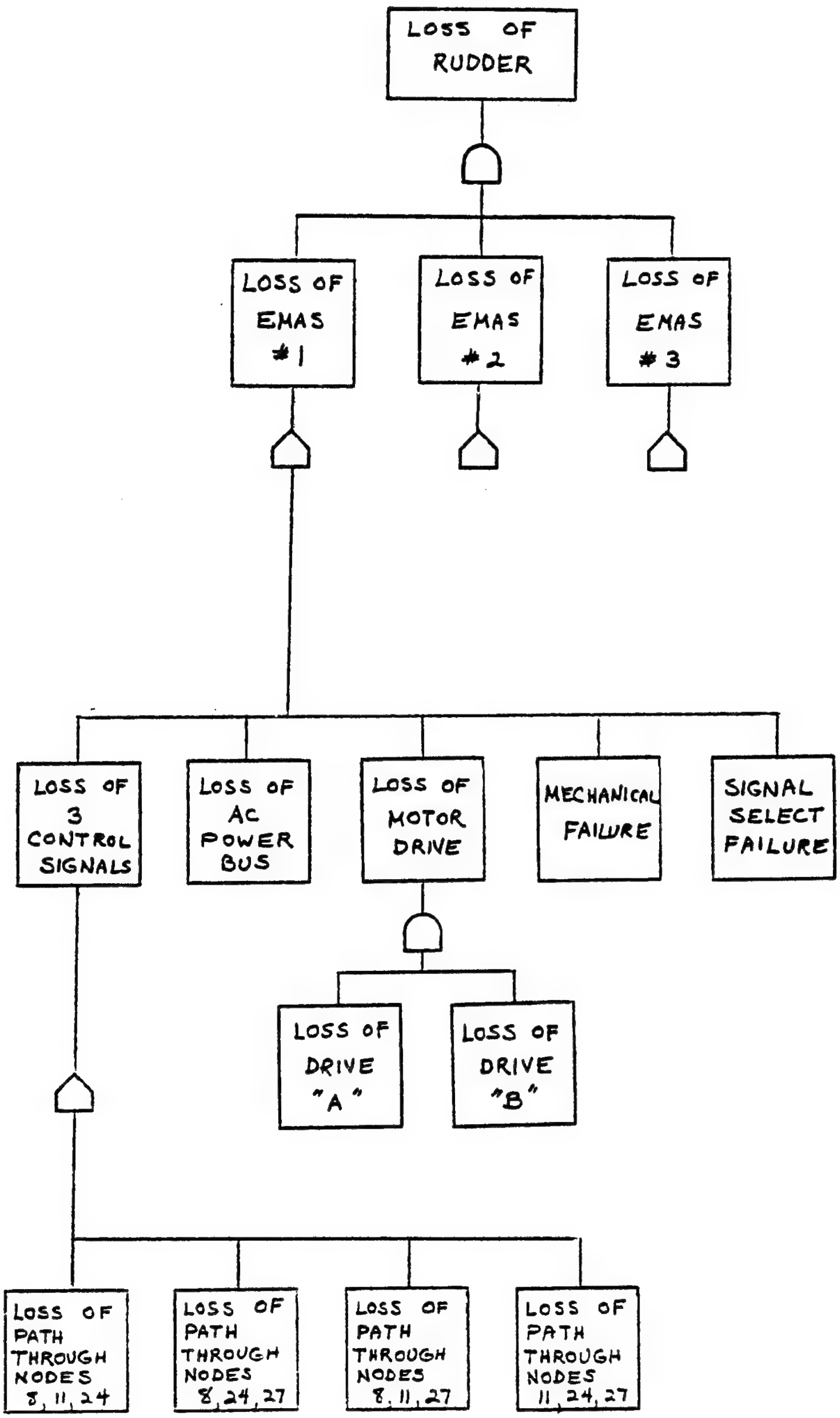
$$M = 2(tt)^2(vv) + 2(tt)(vv)^2 - 3(tt)^2(vv)^2 + D+E+F+G$$

The loss of an inboard, and an outboard aileron, has probability given by:

$$INAIL = M^3 \qquad OUTAIL = M^2$$

The probability of losing roll control is the probability of losing three or more ailerons. The equation for this is:

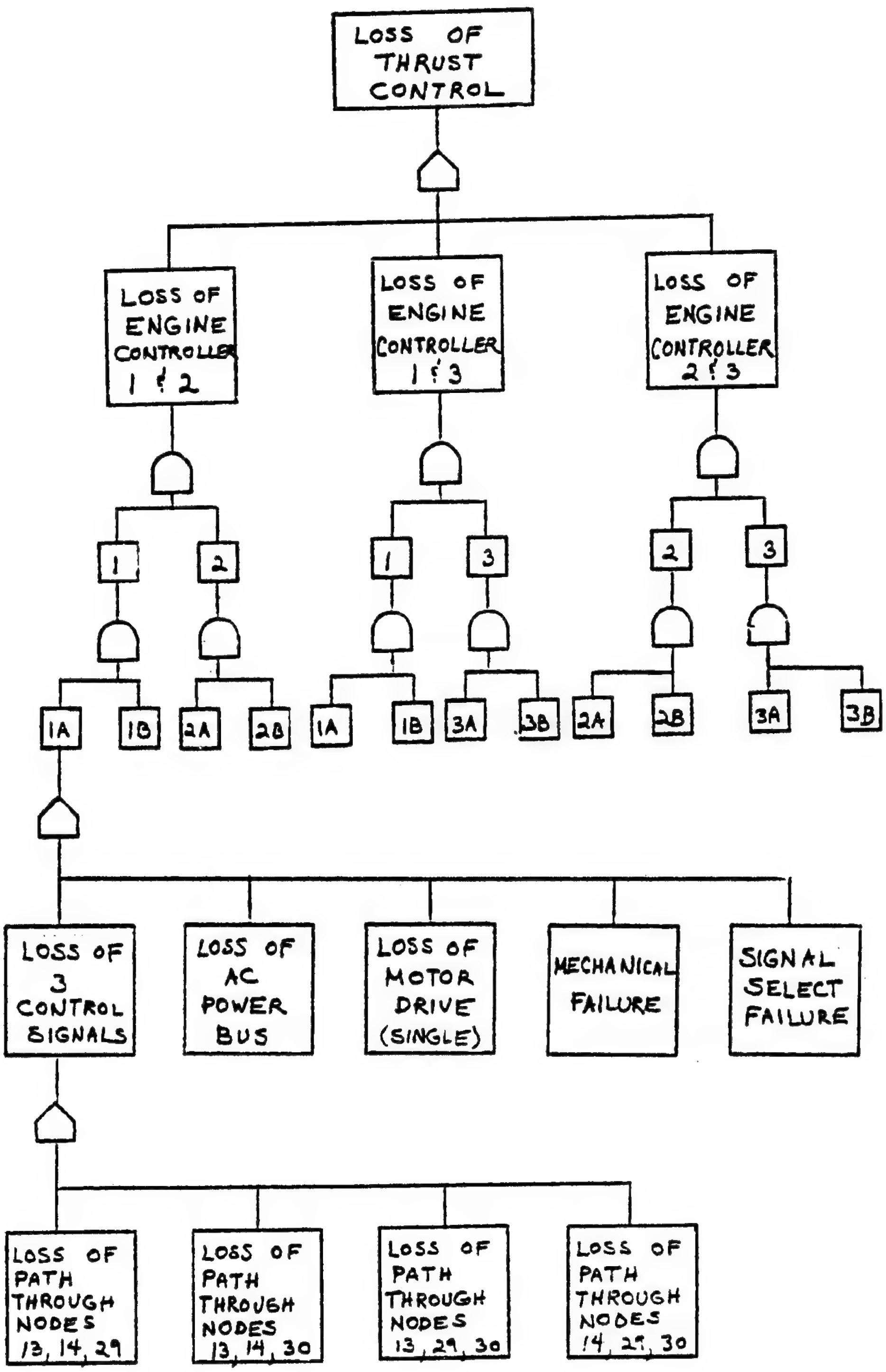
$$
\begin{aligned}
P[\text{3 ailerons lost}] = {} & IN^2 \cdot OUT + IN^2 \cdot OUT + IN \cdot OUT^2 + IN \cdot OUT^2 \\
& - IN^2OUT^2 - IN^2OUT^2 - IN^2OUT^2 - IN^2OUT^2 - IN^2OUT^2 - IN^2OUT^2 \\
& + IN^2OUT^2 + IN^2OUT^2 + IN^2OUT^2 + IN^2OUT^2 \\
& - IN^2OUT^2 \\
= {} & 4M^5 - 3M^6
\end{aligned}
$$


LOSS OF RUDDER
LOSS OF EMAS #1
LOSS OF EMAS #2
LOSS OF EMAS #3
LOSS OF 3 CONTROL SIGNALS
LOSS OF AC POWER BUS
LOSS OF MOTOR DRIVE
MECHANICAL FAILURE
SIGNAL SELECT FAILURE
LOSS OF DRIVE "A"
LOSS OF DRIVE "B"
LOSS OF PATH THROUGH NODES 8, 11, 24
LOSS OF PATH THROUGH NODES 8, 24, 27
LOSS OF PATH THROUGH NODES 8, 11, 27
LOSS OF PATH THROUGH NODES 11, 24, 27

The rudder has three actuators, any one of which is capable of providing sufficient yaw control. Two of the actuators subscribe to nodes 8, 11, 24, and 27, and one to nodes 9, 10, 25, and 26.

P[loss of EMAS #1] = S
P[loss of EMAS #2] = T
P[loss of EMAS #3] = S

Thus, P[loss of rudder] = S^2T


LOSS OF THRUST CONTROL
LOSS OF ENGINE CONTROLLER 1 & 2
LOSS OF ENGINE CONTROLLER 1 & 3
LOSS OF ENGINE CONTROLLER 2 & 3
1
2
1
3
2
3
1A
1B
2A
2B
1A
1B
3A
3B
2A
2B
3A
3B
LOSS OF 3 CONTROL SIGNALS
LOSS OF AC POWER BUS
LOSS OF MOTOR DRIVE (SINGLE)
MECHANICAL FAILURE
SIGNAL SELECT FAILURE
LOSS OF PATH THROUGH NODES 13, 14, 29
LOSS OF PATH THROUGH NODES 13, 14, 30
LOSS OF PATH THROUGH NODES 13, 29, 30
LOSS OF PATH THROUGH NODES 14, 29, 30

Two engine controllers must be operative in order to have sufficient thrust control, but each of the three controllers is dual. Unlike the electromechanical actuators, the "motor drive" on the engine controllers is not dual.

Let P[loss of EMAS motor drive] = E
Then P[loss of engine control motor drive] = $\sqrt[4]{E}$
And,

$P[$ loss of dual engine controller #1 $] = (M - E + \sqrt{E})^2 = ENGC1$
$P[$ loss of dual engine controller #2 $] = (T - E + \sqrt{E})^2 = ENGC2$
$P[$ loss of dual engine controller #3 $] = (M - E + \sqrt{E})^2 = ENGC1$

Finally,

$$\begin{aligned} P[\text{loss of thrust control}] &= P(1\cap 2) + P(1\cap 3) + P(2\cap 3) \\ &\quad - P(1\cap 2\cap 3) - P(1\cap 2\cap 3) - P(1\cap 2\cap 3) \\ &\quad + P(1\cap 2\cap 3) \end{aligned}$$

$$\begin{aligned} = &\; ENGC1 \cdot ENGC2 + ENGC1 \cdot ENGC1 + ENGC2 \cdot ENGC1 \\ &- ENGC1^2 \cdot ENGC2 - ENGC1^2 \cdot ENGC2 - ENGC1^2 \cdot ENGC2 + ENGC1^2 \cdot ENGC2 \end{aligned}$$

$$= 2 ENGC1 \cdot ENGC2 + ENGC1^2 - 2 ENGC1^2 \cdot ENGC2$$

This completes computation of failure probabilities for the subscribers of mesh network #1.

LOSS OF REQUIRED SENSORS

There are 10 types of sensors which input data into this flight control system. They are:

SENSOR	QUANTITY	FAILURE PROBABILITY
1. COLUMN POSITION LVDT	8	$p_1' = 1 - \exp(.7t \times 10^{-6})$
2. HOR. STAB. POS. LVDT	16	$p_2' = 1 - \exp(.7t \times 10^{-6})$
3. RUDDER PEDAL LVDT	8	$p_3' = 1 - \exp(.7t \times 10^{-6})$
4. WHEEL RVDT	8	$p_4' = 1 - \exp(.7t \times 10^{-6})$
5. ANGLE OF ATTACK SENSOR	4	$p_5' = 1 - \exp(2.5t \times 10^{-6})$
6. ENGINE THROTTLE LVDT	12	$p_6' = 1 - \exp(.7t \times 10^{-6})$
7. PITCH RATE GYRO	4	$p_7' = 1 - \exp(26t \times 10^{-6})$
8. YAW RATE GYRO	4	$p_8' = 1 - \exp(26t \times 10^{-6})$
9. NORMAL ACCELEROMETER	4	$p_9' = 1 - \exp(74t \times 10^{-6})$
10. LONGITUDINAL ACCELEROMETER	4	$p_{10}' = 1 - \exp(74t \times 10^{-6})$
11. INERTIAL REFERENCE UNIT	4	$p_{11}' = 1 - \exp(100t \times 10^{-6})$

At least two of each sensor type is necessary to prevent stage failure. Stages 7, 8, 9, 10 are all backed up by the inertial reference unit stage, thus providing some inter-stage redundancy. Let $p_{12} = 1 - \exp(-.075t \times 10^{-6})$ be the failure probability for an avionics power supply. When one power supply fails, one fourth of the sensors of each type fails, also.

In addition to the probability of sensor equipment failure, one must consider the probability that the paths from the node receiving the sensor input to the computer fail. This would be a difficult problem, had it not already been solved for the EMAS control signals. Since these failure probabilities are so close to being the same, regardless of the node or the network under consideration, a constant value may be added to the equipment failure probabilities to account for path failure.

Let $\psi = \frac{ww + xx + yy + zz}{4}$

Then $p_i = p_i' + \psi$, for $i = 1, 2 \cdots$

In the case of network 1, the path failure probability from a node such as node 15 is distinct from that from nodes 8, 9, 10 and 11, and additional values, uu, ss, tt, and vv, were determined. Thus, for network 1 only,

$$\psi = \frac{ww + xx + yy + zz + uu + ss + tt + vv}{8}$$

Let events AOA, HSP, CP, WHL, PED, THR, NA, LA, PRG, YRG, and IRU be such that:

AOA = loss of 3 or 4 angle of attack sensors
HSP = loss of 15 or 16 horizontal stabilyer position LVDT's
CP = loss of 7 or 8 column position LVDT's
WHL = loss of 7 or 8 wheel RVDT's
PED = loss of 7 or 8 rudder pedal LVDT's
THR = loss of engine throttle stage
NA = loss of 3 or 4 normal accelerometer
LA = loss of 3 or 4 longitudinal accelerometers
PRG = loss of 3 or 4 pitch rate gyros
YRG = loss of 3 or 4 yaw rate gyros
IRU = loss of 3 or 4 inertial reference unit

P[loss of required sensor type] = P[loss of sensor not backed up by IRU]
+ P[loss of sensor backed up by IRU] P[loss of IRU]

$$= P\{AOA \cup HSP \cup CP \cup WHL \cup PED \cup THR \cup (IRU) \cap (NA \cup LA \cup PRG \cup YRG)\}$$

Let $SENS = \{AOA \cup HSP \cup CP \cup WHL \cup PED \cup THR \cup (IRU) \cap (NA \cup LA \cup PRG \cup YRG)\}$

$$\begin{aligned} P[SENS] = \; & P[SENS \mid \text{NO AVIONIC POWER SUPPLY LOSS}][\text{NO PS LOSS}] \\ & + P[SENS \mid 1 \text{ PS LOSS}]\, P[1 \text{ PS LOSS}] \\ & + P[SENS \mid 2 \text{ PS LOSS}]\, P[2 \text{ PS LOSS}] \\ & + P[SENS \mid 3 \text{ PS LOSS}]\, P[3 \text{ PS LOSS}] \\ & + P[SENS \mid 4 \text{ PS LOSS}]\, P[4 \text{ PS LOSS}] \end{aligned}$$

The input from the throttles should be treated separately. For each engine, 2 out of 4 throttle LVDT's must be operative. Also, 2 out of 3 engines must have a throttle input. Thus, if all power supplies are working,

$$P[\text{engine \#1 having insufficient throttle input}] = 4p_i^3(1-p_i) + p_i^4$$

$P[$ system having insufficient throttle input $]$

$= P[(T1 \cap T2) \cup (T1 \cap T3) \cup (T2 \cap T3)]$ where T2 = insufficient input to engine # 2

$$\begin{aligned} = {} & P[T1 \cap T2] + P[T1 \cap T3] + P[T2 \cap T3] \\ & - P[(T1 \cap T2) \cap (T1 \cap T3)] - P[(T1 \cap T2) \cap (T2 \cap T3)] - P[(T1 \cap T3) \cap (T2 \cap T3)] \\ & + P[(T1 \cap T2) \cap (T1 \cap T3) \cap (T2 \cap T3)] \end{aligned}$$

$$= 3\left[4p_i^3(1-p_i) + p_i^4\right]^2 - 2\left[4p_i^3(1-p_i) + p_i^4\right]^3 = \text{THR}\emptyset$$

With one power supply failed, this becomes

$$3\left[3p_i^2(1-p_i) + p_i^3\right]^2 - 2\left[3p_i^2(1-p_i) + p_i^3\right]^3 = \text{THR1}$$

With two failed, it is

$$3\left[2p_i(1-p_i) + p_i^2\right]^2 - 2\left[2p_i(1-p_i) + p_i^2\right]^3 = \text{THR2}$$

$$P[\text{SENS}] = P[\text{AOA} \cup \text{HSP} \cup \text{CP} \cup \text{WHL} \cup \text{PED} \cup \text{THR} \cup \text{IRU} \cap (\text{NA} \cup \text{LA} \cup \text{YRG} \cup \text{PRG})]$$

$$\begin{aligned}
= \{ & 8p_1^7(1-p_1) + p_1^8 + 16p_2^{15}(1-p_2) + p_2^{16} + 8p_3^7(1-p_3) + p_3^8 \\
& + 8p_4^7(1-p_4) + p_4^8 + 4p_5^3(1-p_5) + p_5^4 + \text{THR}0 \\
& + [4p_{11}^3(1-p_{11}) + p_{11}^4][4p_7^3(1-p_7) + p_7^4 + 4p_8^3(1-p_8) + p_8^4 \\
& + 4p_9^3(1-p_9) + p_9^4 + 4p_{10}^3(1-p_{10}) + p_{10}^4]\} \quad (1-p_{12})^4
\end{aligned}$$

$$\begin{aligned}
+ \{ & 6p_1^5(1-p_1) + p_1^6 + 12p_2^{11}(1-p_2) + p_2^{12} + 6p_3^5(1-p_3) + p_3^6 \\
& + 6p_4^5(1-p_4) + p_4^6 + 3p_5^2(1-p_5) + p_5^3 + \text{THR}1 \\
& + [3p_{11}^2(1-p_{11}) + p_{11}^3][3p_7^2(1-p_7) + p_7^3 + 3p_8^2(1-p_8) + p_8^3 \\
& + 3p_9^2(1-p_9) + p_9^3 + 3p_{10}^2(1-p_{10}) + p_{10}^3]\} \; 4(1-p_{12})^3 p_{12}
\end{aligned}$$

$$\begin{aligned}
+ \{ & 4p_1^3(1-p_1) + p_1^4 + 8p_2^7(1-p_2) + p_2^8 + 4p_3^3(1-p_3) + p_3^4 \\
& + 4p_4^3(1-p_4) + p_4^4 + 2p_5(1-p_5) + p_5^2 + \text{THR}2 \\
& + [2p_{11}(1-p_{11}) + p_{11}^2][2p_7(1-p_7) + p_7^2 + 2p_8(1-p_8) + p_8^2 \\
& + 2p_9(1-p_9) + p_9^2 + 2p_{10}(1-p_{10}) + p_{10}^2]\} \; 6(1-p_{12})^2 p_{12}^2
\end{aligned}$$

$$+ \{1\} \; 4(1-p_{12})p_{12}^3$$

$$+ \{1\} \; p_{12}^4$$

This formula yields the failure probability for the entire sensor capability. Thus, summing the failure probabilities for sensors, processors, horizontal stabilizers, ailerons, rudder, and thrust control, one has the total flight control failure probability for mesh #1.

MESH NETWORKS #2 AND #3

The analyses for networks 2 and 3 is very similar to the analysis recorded here for network 1, where the probability of a path failure is conditioned on the event of a power failure to each bus controller. The only change is to the values of W, X, Y, and Z, which are shown below.

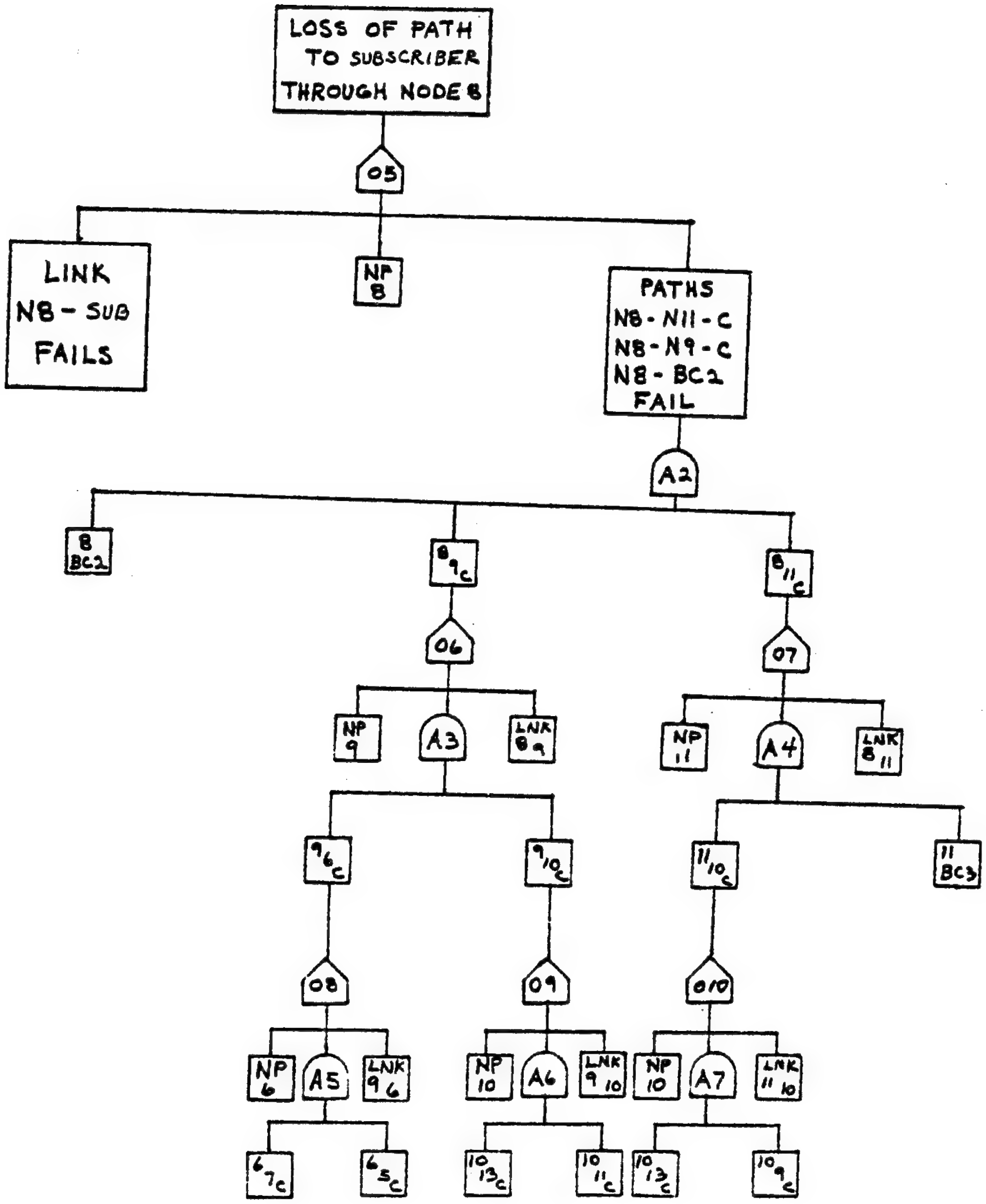
NETWORK 2

$$WW = \mu + \lambda^3 - \theta\mu - \theta\lambda^3 + 2\theta\lambda^2 + \theta = P[EN\ 11, 27, 15, 31, 3, 19, 7, 23]$$
$$XX = \mu + \lambda^3 - \theta\mu - \theta\lambda^3 + 2\theta\lambda^2 + \theta = P[EN\ 8, 24, 12, 28, 0, 16, 4, 20]$$
$$YY = \mu + \lambda^3 - \theta\mu + \theta\lambda^3 + 2\theta\lambda^2 + \theta = P[EN\ 9, 25, 13, 29, 1, 17, 5, 21]$$
$$ZZ = \mu + \lambda^3 - \theta\mu + \theta\lambda^3 + 2\theta\lambda^2 + \theta = P[EN\ 10, 26, 14, 30, 2, 18, 6, 22]$$

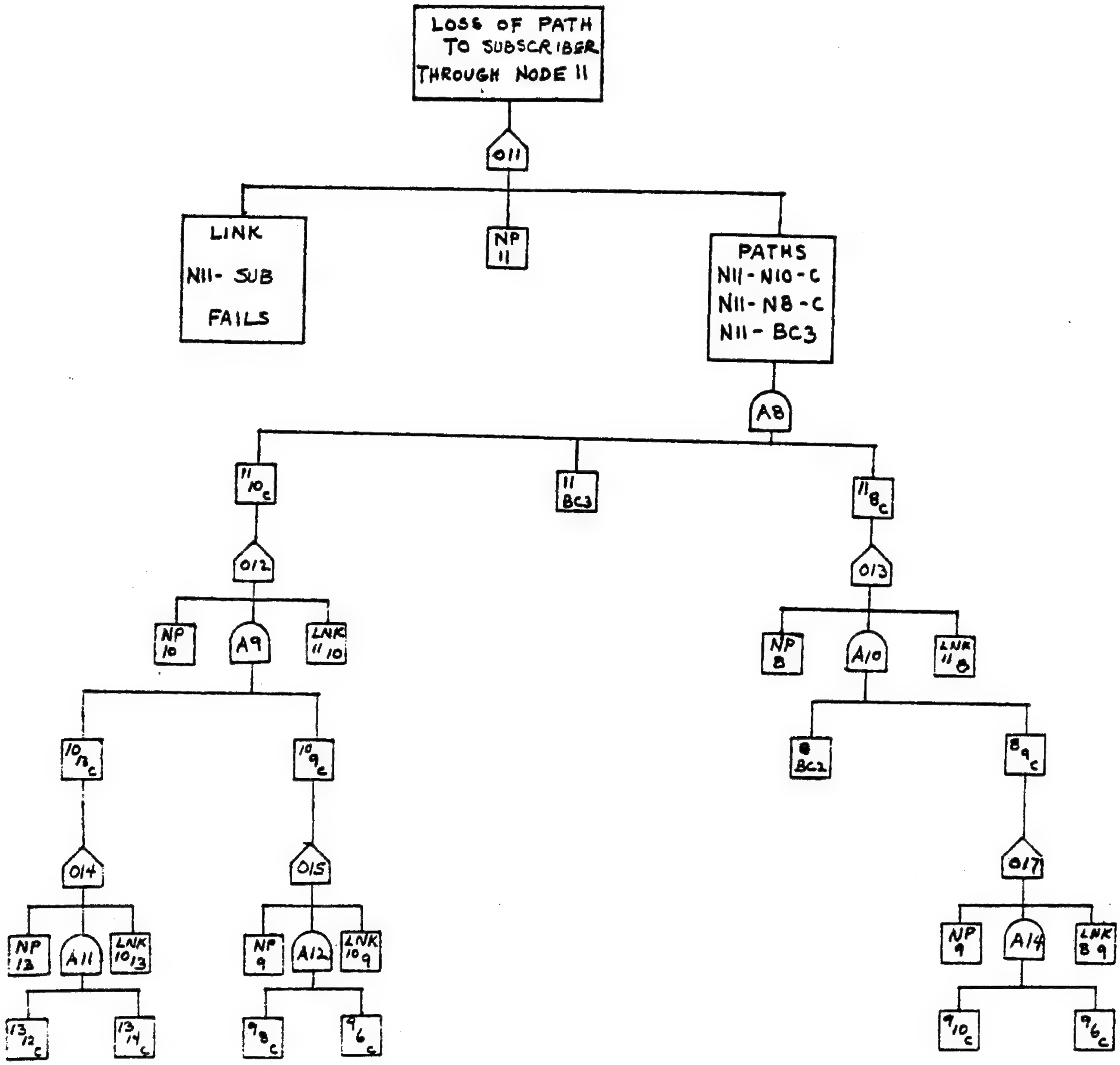
NETWORK 3

$$WW = \mu + 6\lambda^2 + \theta = P[EN\ 11, 27, 15, 31, 3, 19, 7, 23]$$
$$XX = \mu + 6\lambda^2 + \theta = P[EN\ 8, 24, 12, 28, 0, 16, 4, 20]$$
$$YY = \mu + 8\lambda^2 + \theta = P[EN\ 9, 25, 13, 29, 1, 17, 5, 21]$$
$$ZZ = \mu + 8\lambda^2 + \theta = P[EN\ 10, 26, 14, 30, 2, 18, 6, 22]$$

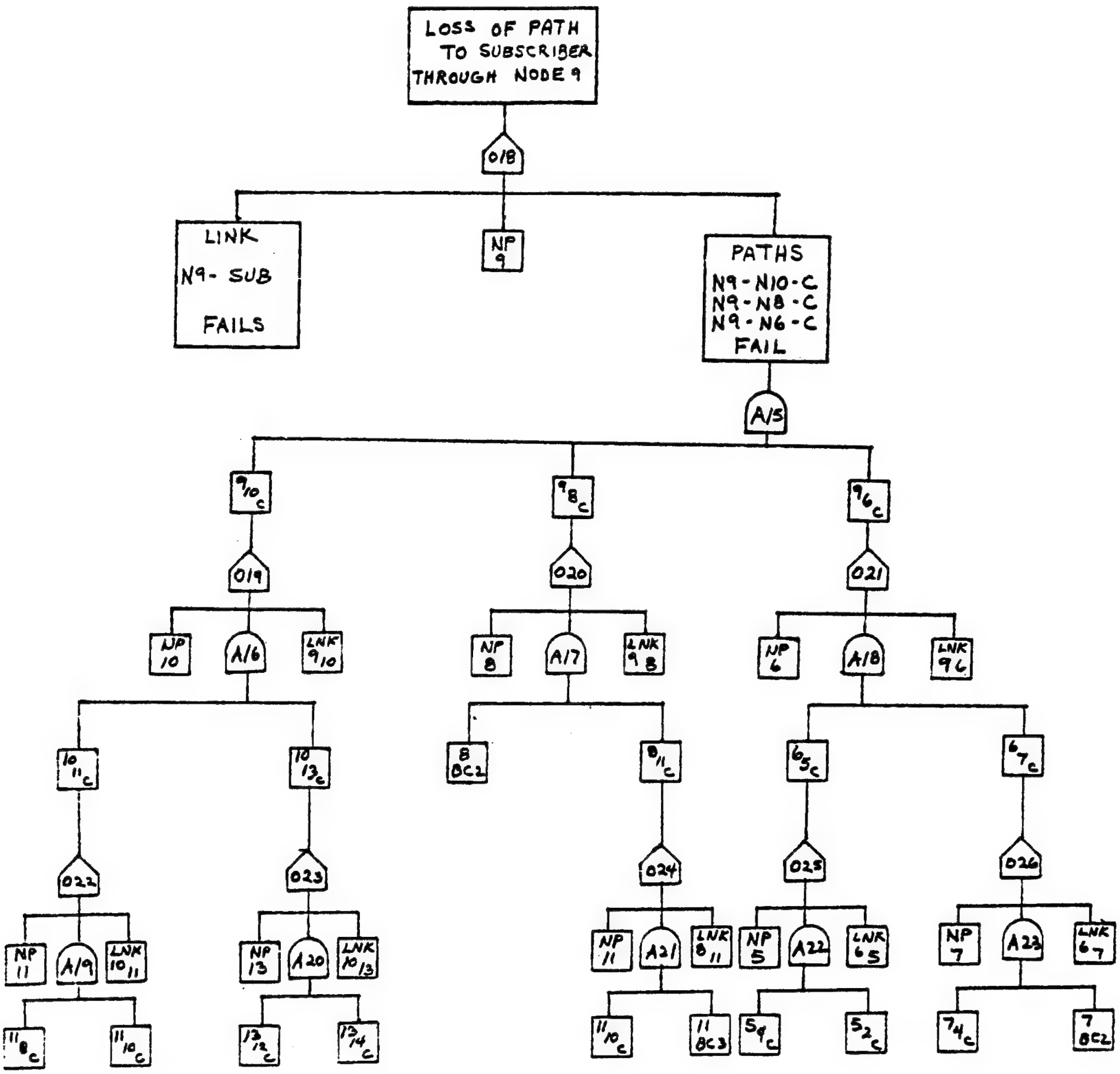
For all three networks, the value of WW, XX, YY, and ZZ are dominated by the terms μ and θ, thus, WW, XX, YY, and ZZ are almost the same for each network. This means that the mesh design has little effect on system reliability, but should influence survivability.


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 8
O5
LINK N8 - SUB FAILS
NP 8
PATHS N8 - N11 - C N8 - N9 - C N8 - BC2 FAIL
A2
8 BC2
A3
A4
NP 9
LNK 8 9
NP 11
LNK 8 11
11 BC3
O6
O7
O8
O9
O10
NP 6
A5
LNK 9 6
NP 10
A6
LNK 9 10
NP 10
A7
LNK 11 10

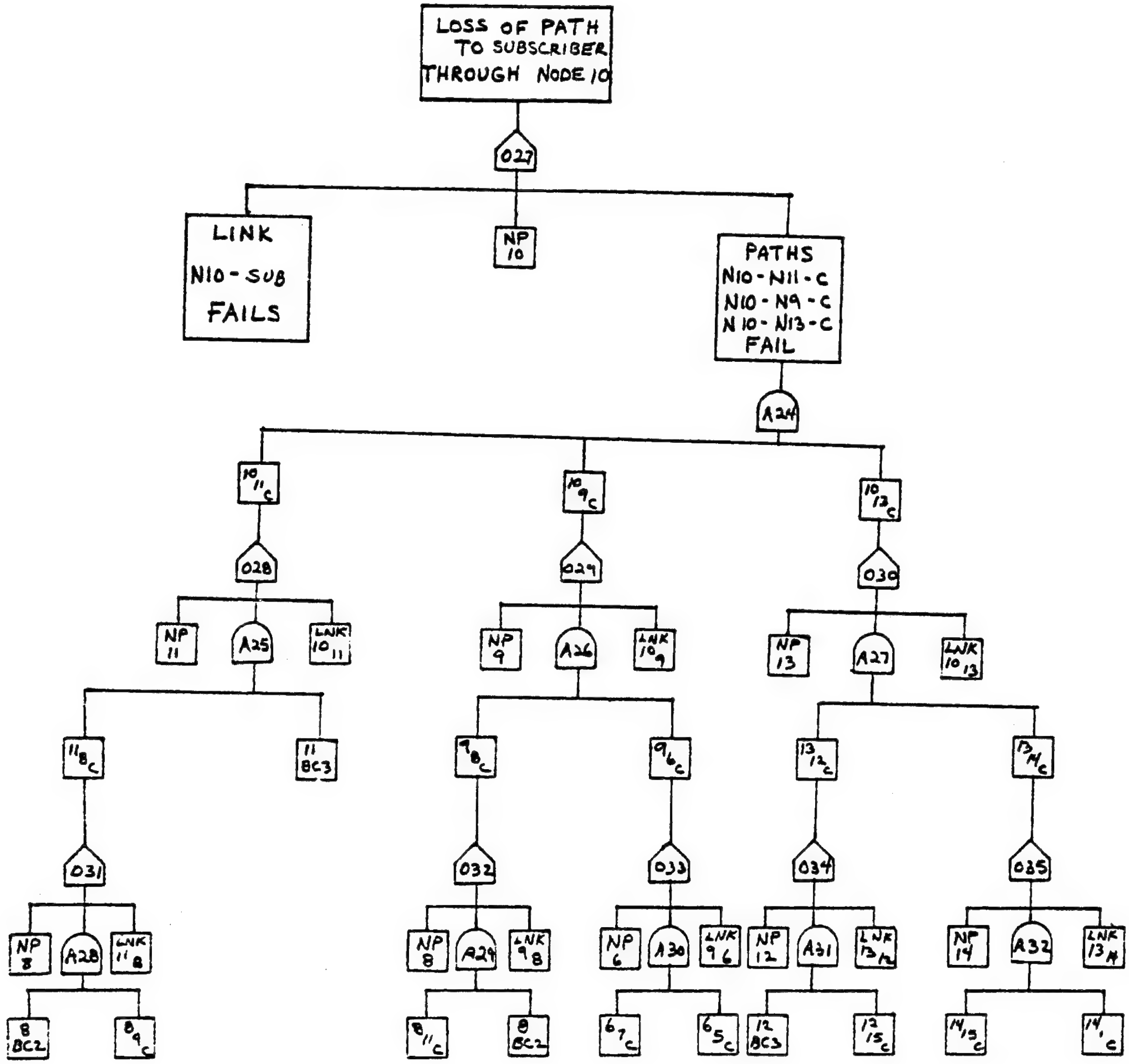
NETWORK 2


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 11
O11
LINK N11- SUB FAILS
NP 11
PATHS N11-N10-C N11-N8-C N11-BC3
A8
11 10 c
11 BC3
11 8 c
O12
O13
NP 10
A9
LNK 11 10
NP 8
A10
LNK 11 8
10 13 c
10 9 c
8 BC2
8 9 c
O14
O15
O17
NP 13
A11
LNK 10 13
NP 9
A12
LNK 10 9
NP 9
A14
LNK 8 9
13 12 c
13 14 c
9 8 c
9 6 c
9 10 c
9 6 c

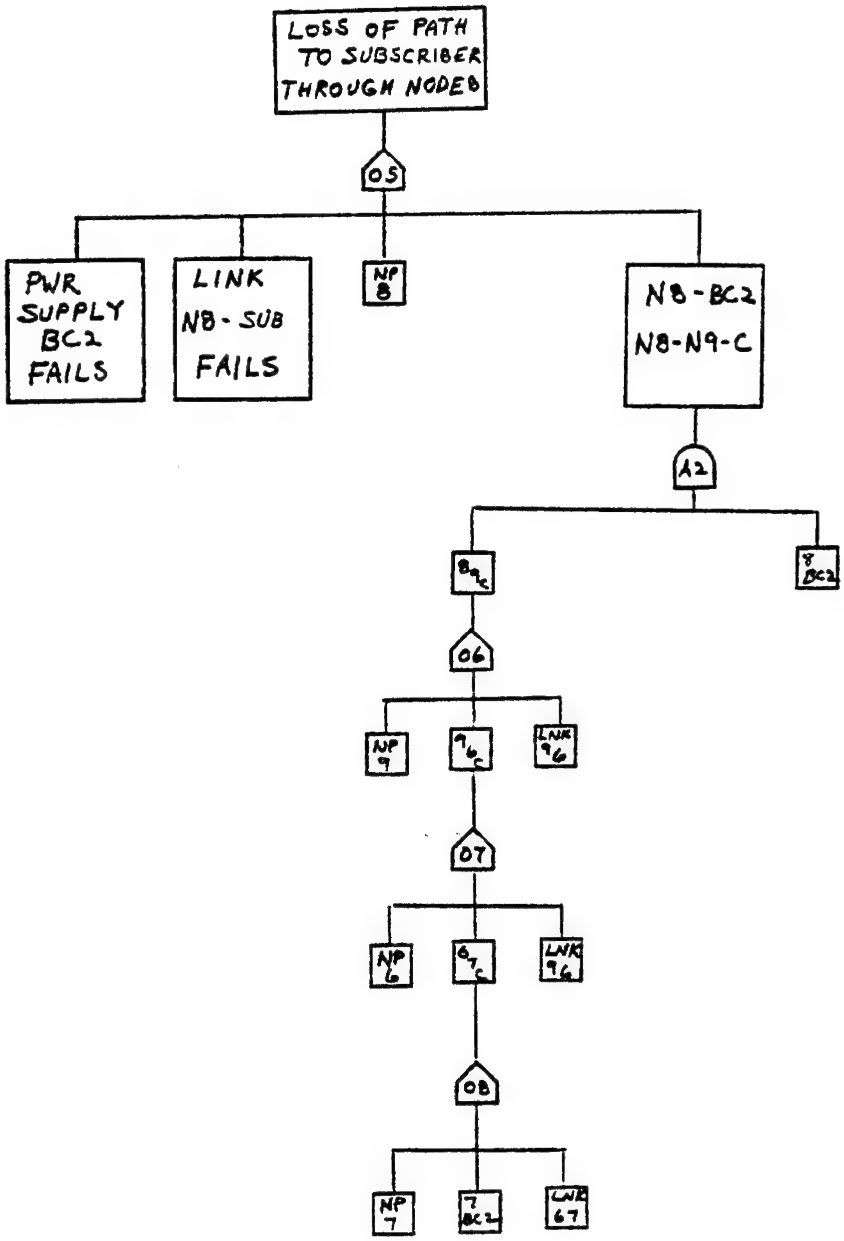
NETWORK 2


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 9
LINK N9- SUB FAILS
NP 9
PATHS N9-N10-C N9-N8-C N9-N6-C FAIL

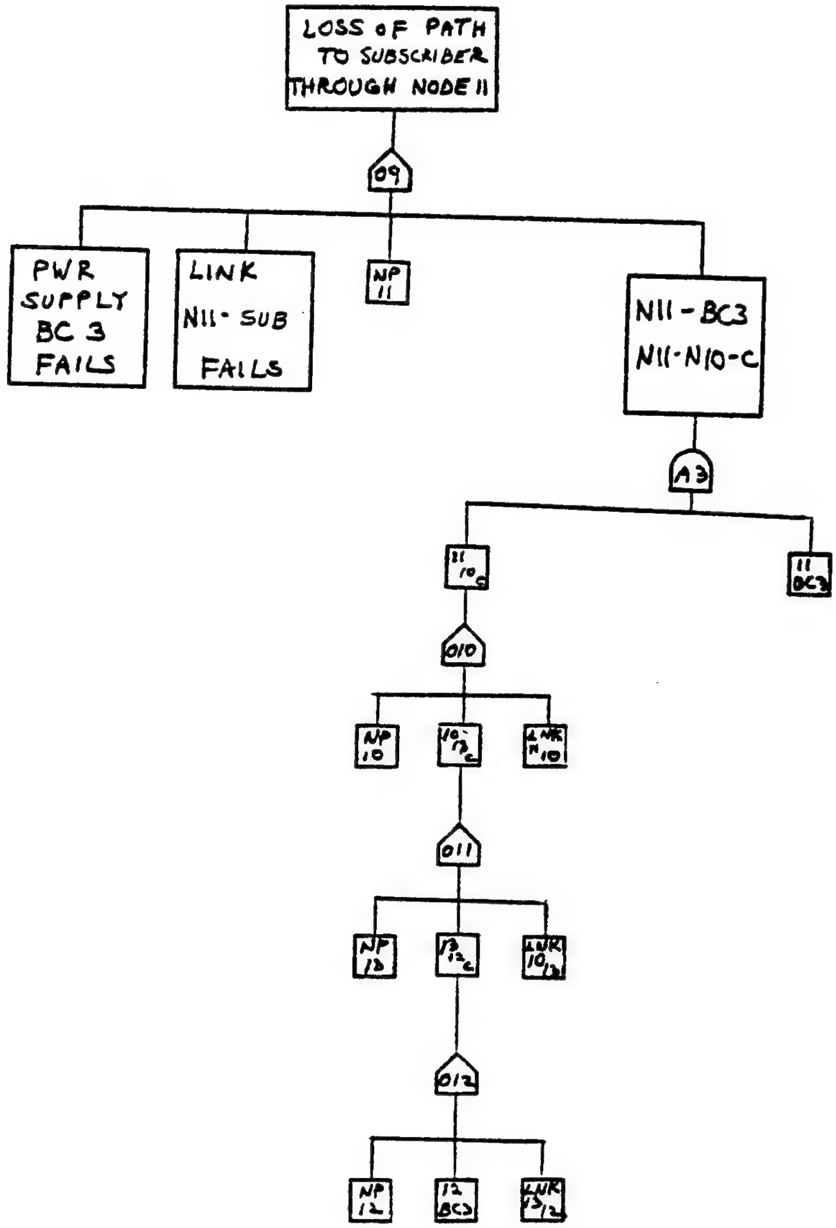
NETWORK 2


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 10
O27
LINK N10 - SUB FAILS
NP 10
PATHS N10 - N11 - C N10 - N9 - C N10 - N13 - C FAIL
A24
O28
NP 11
A25
LNK 10 11
11 BC3
O29
NP 9
A26
LNK 10 9
O30
NP 13
A27
LNK 10 13
O31
NP 8
A28
LNK 11 8
8 BC2
O32
NP 8
A29
LNK 9 8
8 BC2
O33
NP 6
A30
LNK 9 6
O34
NP 12
A31
LNK 13 12
12 BC3
O35
NP 14
A32
LNK 13 14

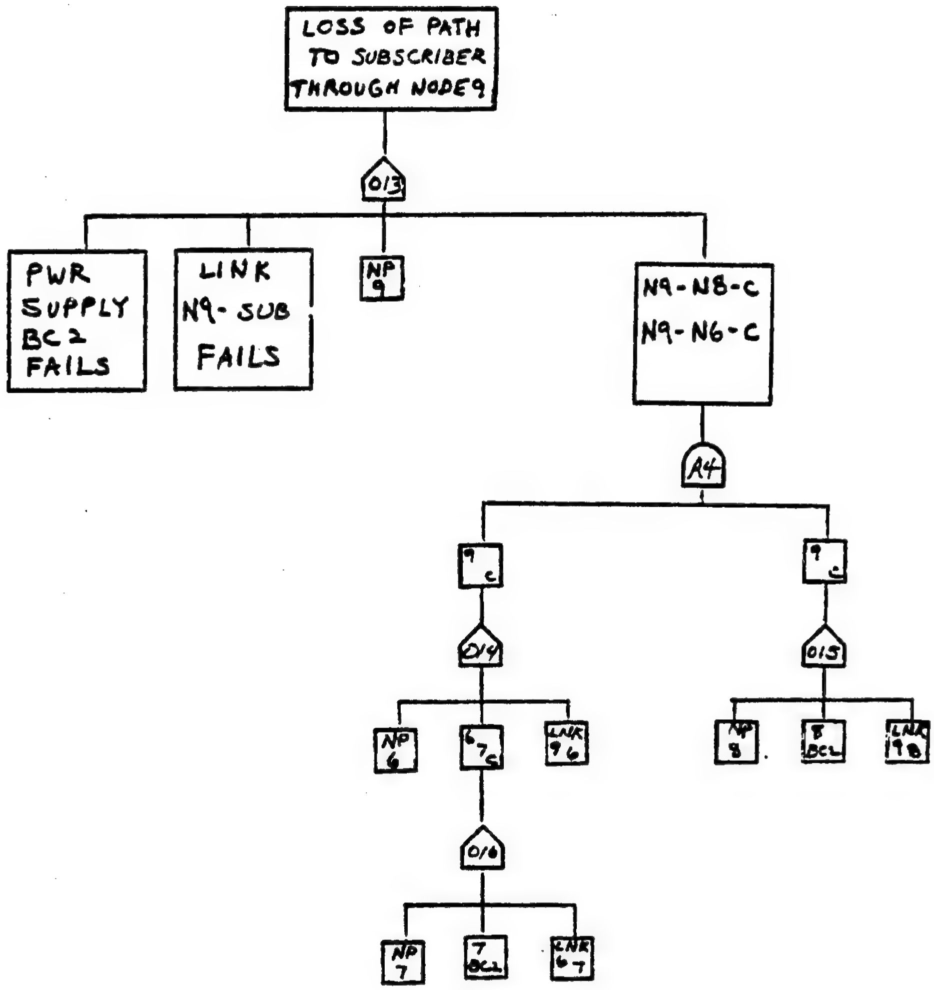
NETWORK 2


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 8
O5
PWR SUPPLY BC2 FAILS
LINK N8-SUB FAILS
NP 8
N8-BC2 N8-N9-C
A2
8 BC2
O6
NP 9
LNK 96
O7
NP 6
LNK 96
O8
NP 7
7 BC2
LNK 67

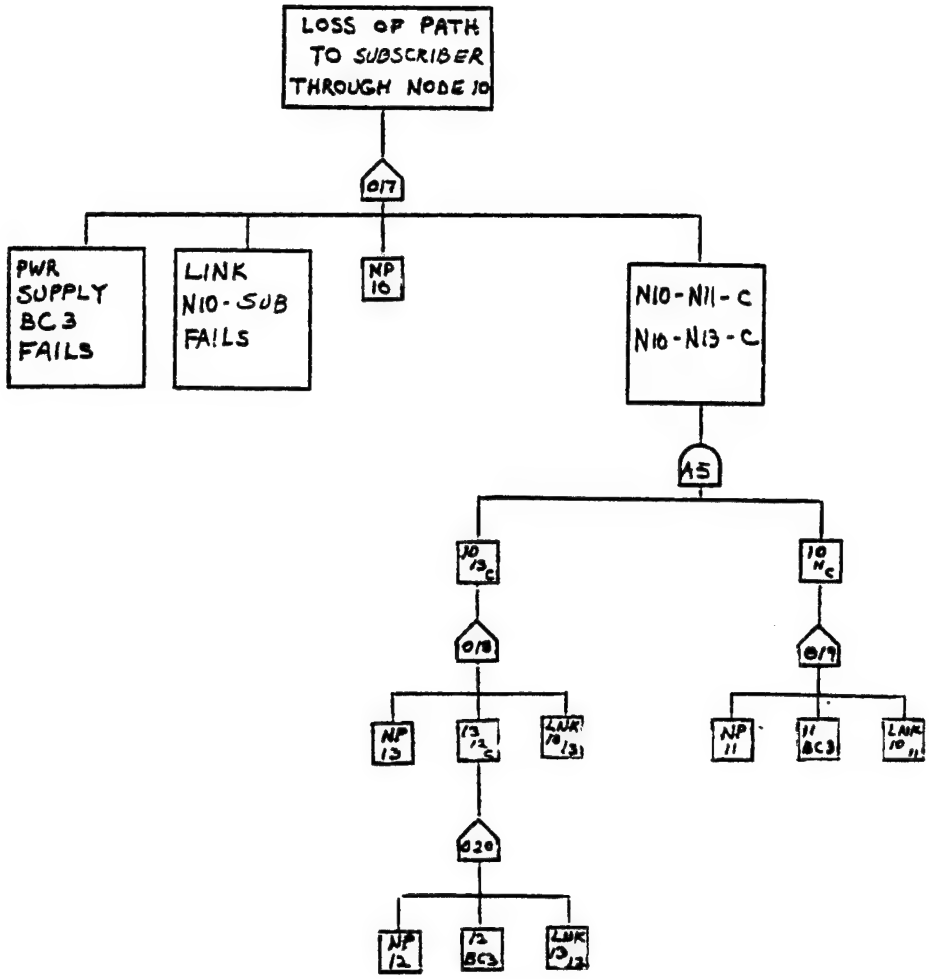
NETWORK 3


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 11
O9
PWR SUPPLY BC 3 FAILS
LINK N11-SUB FAILS
NP 11
N11 - BC3
N11-N10-C
A3
11 BC3
O10
NP 10
O11
NP 13
O12
NP 12
12 BC3

NETWORK 3


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 9
013
PWR SUPPLY BC 2 FAILS
LINK N9-SUB FAILS
NP 9
N9-N8-C
N9-N6-C
A4
9 c
9 c
014
015
NP 6
6 7 c
LNK 96
NP 8
8 BC2
LNK 98
016
NP 7
7 BC2
LNK 67

NETWORK 3


LOSS OF PATH TO SUBSCRIBER THROUGH NODE 10
017
PWR SUPPLY BC3 FAILS
LINK N10-SUB FAILS
NP 10
N10-N11-C
N10-N13-C
A5
018
019
NP 13
LNK
NP 11
11 BC3
LNK
020
NP 12
12 BC3
LNK

NETWORK 3

CONCLUSIONS

All the preceding analyses produced formulae suitable for coding into a computer program, which was done. Using this program produced the results shown at the beginning of the report. These results show that the systems, as described, are quite reliable. The differences in reliability, however, are almost indistinguishable between the three different architectures. Thus, any additional path redundancy may contribute additional survivability, but does little to enhance reliability.

```
C              THIS PROGRAM COMPUTES FAULT TREE TYPE ANALYSIS
C              FOR THE PITCH CONTROL SYSTEM ON THE INTEGRATED
C              DIGITAL ELECTRIC AIRCRAFT (I.D.E.A.) - NASA.
C
      DIMENSION  TIME(20)
C     IMPLICIT DOUBLE PRECISION (A-H,P-Z)
C
       READ (5,*) MESH
C
C
C                   READ IN  MISSION TIMES
C
      READ (5,*) N
      READ (5,*) (TIME(I),I=1,N)
C
C              READ IN THROTTLE LVDT, PEDAL LVDT, WHEEL
C              RVDT, ANGLE OF ATTACK, COLUMN POSITION, HORIZONTAL
C              STABILIZER POSITION, INERTIAL REFERENCE UNIT, NORMAL
C              ACCELEROMETER, LONGITUDINAL ACCELEROMETER, PITCH RATE
C              GYRO, YAW RATE GYRO, & AVIONICS PWR SUPPLY FAILURE RATES.
C
      READ (5,*) THRT1,PED1,WHL1,P1,Q1,R1,S1,T1,V1,W1,YRG1,X2
C
C              READ IN SUBSCRIBER LINK, NETWORK NODE, AND BC POWER
C              SUPPLY FAILURE RATES.
C
      READ (5,*) A1,B1,C1
C
C              READ IN AC PWR BUS, EMAS MOTOR DRIVE, MECHANICAL,
C              AND SIGNAL SELECT FAILURE RATES.
C
      READ (5,*) D1,E1,F1,G1
C
C              READ IN FAULT TOLERANT COMPUTER FAILURE RATE.
C
      READ (5,*) FTC1
C
C               CONVERT INPUT VALUES TO UNIT FAILURE PROBABILTIES.
C
      DO 140 I=1,N
        A=1.-EXP(-A1*TIME(I)/10**6)
        B=1.-EXP(-B1*TIME(I)/10**6)
        C=1.-EXP(-C1*TIME(I)/10**6)
        D=1.-EXP(-D1*TIME(I)/10**6)
        E2=1.-EXP(-E1*TIME(I)/10**6)
        E=E2**2
        F=1.-EXP(-F1*TIME(I)/10**6)
        G=1.-EXP(-G1*TIME(I)/10**6)
        FTC2=FTC1/10**6
        FTC3=FTC2/10**6
        FTC4=FTC3*TIME(I)
        FTC=FTC4-(FTC4**2)/2+(FTC4**3)/6
```

```
C
C            CALCULATE MESH NETWORK FAILURE PROBABILITIES.
C
      IF(MESH.EQ.2) GO TO 70
      IF(MESH.EQ.3) GO TO 80
        WW=C*(B**3+B**2+B-A+1.)+B**3+A
        XX=WW-C*B-2*C*B**3+C*B**2
        YY=XX+2*C*B**3
        ZZ=XX+2*C*B-C*B**2
        UU=C+B**3-C*A+3*B**3*C+6*B**2*C+A
        SS=XX-C*B**3
        TT=WW+C*B**3
        VV=WW+C*B**2
        XNET=(WW+XX+YY+ZZ+UU+SS+TT-VV)/8.0
       GO TO 90
C
   70  WW=A+B**3+C*(2*B**2-A-B**3+1.)
       XX=WW
       YY=WW+2*C*B**3
       ZZ=YY
       XNET=(WW+XX+YY+ZZ)/4.0
      GO TO 90
C
   80  WW=A+6*B**2+C
       XX=WW
       YY=A+8*B**2+C
       ZZ=YY
       XNET=(WW+XX+YY+ZZ)/4.0
C
C            CALCULATE EMAS FAILURE PROBABILITIES.
C
   90   EMAS1=2*XX**2*WW+2*XX*WW**2-3*XX**2*WW**2-D+E+F+G
        EMAS2=2*YY**2*ZZ+2*YY*ZZ**2-3*YY**2*W**2+D+E+F+G
C
        HRSTB=2*EMAS1**2*EMAS2+2*EMAS1*EMAS2**2-3*EMAS1**2*EMAS2**2
C
        EMAS3=2*TT**2*VV+2*TT*VV**2-3*TT**2*VV**2+D+E+F+G
         IF(MESH.GT.1) EMAS3=EMAS2
C
        AILLI=EMAS3**3
        AILLO=EMAS3**2
C
        AIL=2*AILLI**2*AILLO+2*AILLI*AILLO**2-3*AILLI**2*AILLO**2
C
         RUD=EMAS1*EMAS1*EMAS2
        ENGC1=(EMAS3-E+E2)**2
        ENGC2=(EMAS2-E+E2)**2
C
        THRCTL=2*ENGC1*ENGC2+ENGC1**2-2*ENGC1**2*ENGC2
C
```

```
C
C            SENSOR EQUIPMENT FAILURE RATES
C
       THRT=1.-EXP(-THRT1*TIME(I)/10**6)+XNET
       PED= 1.-EXP(-PED1*TIME(I)/10**6)+XNET
       WHL= 1.-EXP(-WHL1*TIME(I)/10**6)+XNET
       P=1.-EXP(-P1*TIME(I)/10**6)+XNET
       Q=1.-EXP(-Q1*TIME(I)/10**6)+XNET
       R=1.-EXP(-R1*TIME(I)/10**6)+XNET
       S=1.-EXP(-S1*TIME(I)/10**6)+XNET
       T=1.-EXP(-T1*TIME(I)/10**6)+XNET
       V=1.-EXP(-V1*TIME(I)/10**6)+XNET
       W=1.-EXP(-W1*TIME(I)/10**6)+XNET
       YRG= 1.-EXP(-YRG1*TIME(I)/10**6)+XNET
       X=1.-EXP(-X2*TIME(I)/10**6)
       X1=1.-X
C
C            CALCULATE LOSS OF SUFFICIENT SENSOR INPUT.
C
       TERM1=P**3*(4-3*P)+Q**7*(8-7*Q)+R**15*(16-15*R)
       TERM15=PED**7*(8-7*PED)+WHL**7*(8-7*WHL)
       TERM2=S**3*(4-3*S)
       TERM3=T**3*(4-3*T)+YRG**2*(4-3*YRG)+V**3*(4-3*V)+W**3*(4-3*W)
       TERM4=P**2*(3-2*P)+Q**5*(6-5*Q)+R**2*(3-2*R)
       TERM45=PED**5*(6-5*PED)+WHL**5*(6-5*WHL)
       TERM5=S**2*(3-2*S)
       TERM6=T**2*(3-2*T)+V**2*(3-2*V)+W**2*(3-2*W)+YRG**2*(3-2*YRG)
       TERM7=P*(2-P)+Q**3*(4-3*Q)+R*(2-R)
       TERM75=PED**3*(4-3*PED)+WHL**3*(4-3*PED)
       TERM8=S*(2-S)
       TERM9=T*(2-T)+V*(2-V)+W*(2-W)+YRG*(2-YRG)
       TERM10=3*((4-3*THRT)*THRT**3)**2-2*((4-3*THRT)*THRT**3)**3
       TERM11=3*((3-2*THRT)*THRT**2)**2-2*((3-2*THRT)*THRT**2)**3
       TERM12=3*((2-THRT)*THRT)**2-2*((2-THRT)*THRT)**3
C
       PART1=X1**4*(TERM15+TERM10+TERM1+TERM2*TERM3)
       PART2=4*X*X1**3*(TERM45+TERM11+TERM4+TERM5*TERM6)
       PART3=6*X**2*X1**2*(TERM75+TERM12+TERM7+TERM8*TERM9)
       PART4=4*X**3*X1
       PART5=X**4
C
       SENS=PART1+PART2+PART3+PART4+PART5
C
       WRITE (6,100) TIME(I)
 100   FORMAT (///1X,'TIME= :',F5.2)
       WRITE (6,110) SENS
 110   FORMAT (1X,'SENSOR FAILURE PROBABILITY: ',E12.5)
C
C            CALCULATE TOTAL FAILURE PROBABILITY.
C
       SYST=FTC+SENS+HRSTB+AIL+RUD+THRCTL
```

```
C
       WRITE (6,120) FTC,SENS,HRSTB,AIL,RUD,THRCTL
  120 FORMAT (1X,'INTERMEDIATE VALUES:   ',6(E9.4,1X))
      WRITE (6,130) SYST
  130 FORMAT (//1X,'FLIGHT CONTROL SYSTEM FAILURE PROB.:   ',E12.7)
  140 CONTINUE
C
      WRITE(6,145) WW,XX,YY,ZZ,UU,SS,TT,VV
145   FORMAT(//1X,8(E9.4,1X))
      WRITE (6,150) P,Q,R,S,T,V,W,X
  150 FORMAT (///1X,'ECHO:  ',8(E9.4,1X))
      WRITE (6,160) A,B,C,D,E,F,G,FTC
  160 FORMAT (//1X,'ECHO:   ',8(E9.4,1X)///)
C
      STOP
      END
```

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16. Abstract

The Integrated Digital/Electric Aircraft (IDEA) is an aircraft concept which employs all-electric secondary power systems and advanced digital flight control systems; these technologies would reach "technology readiness" in 1990. After trade analysis, preferred systems were applied to the baseline configuration. The baseline carries 350 passengers over a design range of 4600 n.mi. An additional configuration, the Alternate IDEA, was also considered. For this concept the design ground rules were relaxed in order to quantify additional synergistic benefits. The major objectives of the study were to define an IDEA configuration, define technical risks associated with the IDEA systems concepts, and to identify the research and development required activities to reduce these risks. The selected subsystems include: power generation - two samarium cobalt starter/generators per engine, producing variable voltage/variable frequency with selective conversion (to 270 Vdc or to constant voltage constant frequency); power distribution - hybrid ring/distributed bus; actuators - rotary power hinge operating through an antijam mechanism; environmental control system - electrically driven vapor cycle system with power regeneration using non-recirculated air; flight controls systems - centralized/mesh architecture, (system reliability permits -10 percent static stability margin). When the aircraft was resized, block fuel was predicted to decrease by 11.3 percent, with 7.9 percent decrease in direct operating cost. The Alternate IDEA shows a further 3.4 percent reduction in block fuel and 3.1 percent reduction in direct operating cost, primarily as a result of reduction in maneuver load factor from 2.5 to 2.0. Technical development is required in power generation, motor technology, ECS concepts, solid-state power controllers, digital data management and flight control systems. Lightning protection also represents significant risk, particularly with respect to composite aircraft.

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